GSFC-S-480-25.1

PERFORMANCE SPECIFICATION

FOR THE

NOAA-K, L, M, N, & N-prime

SATELLITES

GSFC S-480-25 Issued July 1988 for NOAA-KLM

GSFC S-480-25.1 Issued December 1994 for NOAA-KLMNN-prime

GODDARD SPACE FLIGHT CENTER
GREENBELT, MD

REVISION HISTORY

MOD NO.	DATE	SECTION
KLM Contract Award	07/26/88	Initial Release S-480-25
11	05/30/89	2.1, 3.6.1.6
17	09/18/89	3.2.5.1.c, 4.2.1.3, 4.4.4
20	10/18/89	3.6.6 through 3.6.6.4, 3.6.6.5
21	10/17/89	2.5
22	11/22/89	3.6.1.6g
24	12/07/89	2.5
26	03/26/90	3.6.1.6g
28	06/13/90	2.5
30	05/17/90	3.6.1.8
34	08/14/90	2.5
42	02/01/91	3.2.1.2
43	01/29/91	3.6.1.8
44	03/07/91	Appendix A
47	03/22/91	3.6.1.8
48	04/08/91	Cancels Mods 30, 43, 47; 3.6.1.8
54	08/14/91	3.6.1.8
59	09/17/91	Figure 18
70	06/29/92	3.6.4.1.3, 6.4.2
88/131	08/02/93 12/09/94	Tables 3 & 4, 3.2.2.1, 3.2.5.1, 3.6.1.1, 4.2.1.3, 4.2.2.1, 4.4.4
90	05/14/94	2.5, Table 1
93	07/16/93	Figure 19
95	07/16/93	Tables 1, 3, and 14, 3.2.2.1, 3.4.7
96	09/27/93	2.5, Table 1
99	09/27/93	Tables 1, 3, and 4, 3.2.2.1, 3.4.7
110	05/02/94	3.6.1.5
129	12/01/94	4.4.4

REVISION HISTORY (Continued)

MOD No. 132 N/N-prime Mod Award	12/16/94 Section(s) Changed	
Initial Release S-480-25.1	Document Reformatted and Repaginated	
Sections Renumbered	From	To
	3.3.5.1	3.3.5.1.a
	3.3.5.2	3.3.5.1.b
	3.3.5.3	3.3.5.1.c
	3.3.5.4	3.3.5.1.d
	3.3.5.5	3.3.5.1.e
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	3.3.5.13	3.3.6.5
	3.3.5.14	3.3.6.6
	3.3.5.15	3.3.5.2.t
Administrative Changes	2.1, 2.5, 3.1.4.1, 3.1.4.2, 3.2.1.1.a, 3.2.1.1.b, 3.2.1.2, 3.2.2.1.b, 3.2.5.1.c, 3.2.5.1.d, 3.3.1, 3.3.1.1, 3.3.1.2, 3.3.2, 3.3.3, 3.3.5.1, 3.6.1.1.a, 3.6.1.1.b, 3.6.1.4.d, 3.6.1.4.f, 3.6.1.5.g, 3.6.1.5.p, 3.6.1.6.f, 3.6.1.7.b, 3.6.1.7.d, 3.6.1.7.f, 3.6.1.7.h, 3.6.1.7.i, 3.6.1.9, 3.6.1.10, 3.6.4.1, 3.6.5, 3.6.6.4, 3.7, 4.1.4.1, 4.1.5, 4.1.8, 4.1.8.1, 4.2.2.1, 4.4.1.2, Table 1, Table 8 Appendix A: I, IIA, IIB, IIC, IID, III, IV	
N/N-prime Scope Additions:	1.0, 3.6.1.5.a, 3.6.4.1, 3.6.4.2, 3.6.6, 4.2.2, 4.3, 4.4, 4.4.7, Table 2, Table 7, Appendix A: I	
Transfers to Statement of Work	3.2.1.2 (partial), 3.4 (entire section), 3.5 (entire section), 3.6.3.3, 3.6.3.4, 4.1.5	
Transfers to Performance Assurance Requirements, S-480-26.1	3.3.1.1.a, 3.3.1.1.b, 3.3.1.2.a, 3.3.1.2.b, 3.3.1.2.c, 3.3.1.2.d, 3.3.1.2.e, 3.3.2, 3.3.5.16, 3.3.6 (entire section), 3.3.7 (entire section), 3.7.1, 3.7.2, 3.7.3, 3.7.4, 3.7.5	

REVISION HISTORY (Continued)

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135	1249	02/22/95	2.5, 3.6.1.7.c, 3.6.1.7.e, 3.6.1.7.f, Table 1, Table 2, Figure 5, Figure 21, Appendix B
136	1143, 1144	03/01/95	3.2.2.1, 4.2.1.3 -- Definitizes Mod 88/131
140	1313	03/22/95	3.6.1.7.f
143	1277B, 1267, 1279	06/02/95	3.2.5.1.c, Table 4
144	1330A	06/12/95	Document Repaginated plus changes in following: 2.5, 3.1.1.4, 3.1.4.1, 3.1.6, 3.2.1.1a, 3.2.1.1b, 3.2.2.1a, 3.2.2.1b, 3.2.5.1c, 3.6.1.1, 3.6.1.1a, 3.6.1.1b, 3.6.1.2k, 3.6.1.2l, 3.6.1.3a, 3.6.1.3c, 3.6.1.3.d, 3.6.1.8a, 3.6.1.8b, 3.6.1.8f, 3.6.1.8h, 3.6.1.9, 3.6.3.1, 3.6.5, 4.4.4, 4.4.5, 4.4.7, Table 1, Table 3a, Table 3b, Table 4a, Table 4b, Figure 6a, Figure 6b
149	1346	08/16/95	3.2.5.1c, 4.4.4
150	1337	08/16/95	Appendix A
151	1349	08/18/95	3.6.1.6
152	1347	08/22/95	Administrative changes: 3.6.1.2h, 3.6.1.3.d, 3.6.1.8, 3.6.1.9, 3.6.2.3.b, Table 13
156	1363	09/25/95	3.6.1.2k, 3.6.1.8b, 3.6.1.8f, 3.6.1.8h, 3.6.2.3b, Figure 22a, Figure 22b, Table 13
169	1303, 1304, 1380, 1391	02/14/96	3.6.1.5g
170	2306, 1391, 1388, 1364	02/27/96	Administrative changes: 3.2.5.1c, 3.6.1.5g, 3.6.1.6b, 3.6.1.6g, 3.6.1.6h, 4.4.4, Figure 20
174	1367A	04/12/96	2.5
183	2324	07/11/96	3.6.1.3c
184	1388, 1346	07/11/96	3.2.5.1c, 4.4.4
188	1139B	08/15/96	3.6.1.5j, Table 11
191	2302, 1364, 1298, 1298A, 1365, 1396, 1262	10/09/96	Table 2
204	1433, 1437, 1439	12/20/96	3.6.1.6b, 3.6.1.6g, 4.4.7
213	1464A	02/24/97	3.2.1.1
226	1427	06/12/97	3.6.1.7, Table 1, Table 2,, Figure 21a, Figure 21b, Appendix B
229	1432	06/25/97	3.2.5.1, 4.4.4, 4.4.4.1, 4.4.4.2, 4.4.4.3
227	2327	07/23/97	2.1, 3.6.1.6
250	1508	10/27/97	3.6.1.6g
292	1542A	06/09/98	Table 5, Table 9, Table 10
300	1542A	08/24/98	Table 5, Table 9, Table 10
320	1471A	02/03/99	3.2.1.1(b)
328	1436	06/08/99	4.4.6, 4.4.11
331	1603C	07/15/99	2.5, Table 1, Table 2, 3.6.1.5j, 3.6.1.5r
333	1633	07/28/99	3.1.4.3, 3.6.1.7b,c,g, Table 1, Table 2, Figure 21a
338	-	08/09/99	Table 2

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365	02/17/00	2390	02/17/00	4.4.1.1, 4.4.1.2
381	04/20/00	1650	04/20/00	3.6.1.5e
386	05/04/00	1689	05/04/00	2.5, Figure 5, Table 1
391	06/01/00	-	06/01/00	Appendix C (new)
411	09/26/00	2404	09/26/00	3.2.2.1, Table 3a
426	02/12/01	1594B	02/13/01	2.5
427	03/20/01	1704	02/21/00	4.4.1.2
428	03/20/01	1787A, 1732	04/05/01	3.6.1.1, Add paragraph
445	07/16/01	2413	06/12/01	Section 2.4
451	03/08/01	1855	08/07/01	Pages 3, 9, 12, 16, 21, 35, 37, 100
481	1/30/02	1891A	01/07/02	2.5, 3.1.4.1, Table 1, 3.2.1.1, 3.2.2.1, 3.6.1.1, 3.6.5
500	04/19/02	1808	03/25/02	Appendix A
505	05/16/02	2387A	06/11/00	Appendix C
505	05/16/02	2400	07/10/00	Appendix C
527	10/11/02	1943	09/23/02	Section 1
548	03/14/03	1948R1	02/24/03	Section 2.5
548	03/14/03	1949R1	02/25/03	Section 2.5
550	03/19/03	2411BR3	03/04/03	Appendix C
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566	08/15/03	2422R1	08/11/03	4.4.7.1, 4.4.7.2, 4.4.8.1, Table 13
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1 SCOPE

This specification establishes the performance, design, development, and test requirements for the ATN follow-on NOAA-K through -N-prime satellites.[1]

The overall intent of this specification is to provide a spacecraft capable of producing sensor data consistent with nominal instrument performance. The specifications included herein are system requirements measured on a complete satellite configuration. All effects, including mutual conductance and radiative interferences, between satellite subsystems, instruments, harnesses, and tape recorders shall be considered.

It is also intended that spacecraft subsystem operation shall not degrade instrument performance and that the satellite systems produce high quality data that is free from extraneous noise and interference. Signal-to-noise requirements are expressed in terms of random statistics; however, it is the intent that coherent noise components be reduced to a level where they do not appear in the data products. To meet this end, the contractor shall use the best engineering practices in harness, design, and decoupling filter techniques.

The National Oceanic and Atmospheric Administration series of polar orbiting satellites

2 APPLICABLE DOCUMENTS

The latest issue of the following documents forms a part of this specification to the extent specified herein. In the event of conflict between the document and this specification, the contents of this specification shall be considered a superseding requirement.

2.1 SPECIFICATIONS

GSFC S-480-26.1	Performance Assurance Requirements for the NOAA-K, L, M, N, N-prime Satellites, November 1994
WSMCR 127-1	Range Safety Requirements, Range Safety Regulation, May 15, 1985
X-600-87-11	Metsat Charged Particle Environment Study, August 1987 (Revised)
MIL-F-7179E	Finishes and Coatings; Protection of Aerospace Weapons Systems, Structures, and Parts; General Specification for, November 1972
MIL-P-26536	Propellant, Hydrazine
MIL-P-27401	Nitrogen Pressurant

2.2 STANDARDS

FED-STD-209B	Federal Standard, Clean Room and Work Station Requirements, Controlled Environment, April 1973
GSFC-X-560-63-3	Aerospace Data System Standard, July 1971
MIL-STD-454E	Standard General Requirement for Electronic Equipment, 1 March 1976
DOD-STD-480B	Configuration Control - Engineering Changes, Deviations and Waivers
MIL-STD-810	Environmental Test Methods for Aerospace Equipment, 15 June 1967
MIL-STD-882A	System Safety Program Requirements, June 28, 1977
MIL-STD-1574A	System Safety Program for Space and Missile Systems, August 15, 1979

2.3 PUBLICATIONS

MIL HDBK-5	Metallic Material and Elements for Aerospace Vehicle Structures

2.4 FORMS

GSFC-480-39A	Metsat Configuration Change Request Form

2.5 OTHER DOCUMENTS

PPL-18	GSFC Preferred Parts List, Oct. 1986	
530-UGD-GN	Ground Network (GN) Users' Guide, June 1993	
ICD-TII-25004	Interface Control Document to the Advanced Tiros-N Spacecraft (NOAA-K, L, M)	Mod 481 CCR 1891A
MDC 00H0072	National Oceanic and Atmospheric Administration – N (NOAA-N) Mission Specification October 2001.	
ICD-DII-tbd	Interface Control Document to the Advanced Tiros-N Spacecraft (NOAA-N, N-prime)	
E 6.12	Controlled Clean Areas	
2280728	Process Standard Operating Instruction Controlled Clean Area	
2280887	Process Standard Cleaning at Spacecraft Level	
IS-3267415	ATN-KLM General Instrument Interface Specification	
IS-20029950	Unique Interface Specification for the Advanced Very High Resolution Radiometer (AVHRR-3)	
IS-3267402	Unique Interface Specification for the Data Collection and Location System (DCS-2)	Mod 548 CCR 1948R1
IS-23033279	Unique Instrument Interface Specification for the Advanced Data Collection and Location System (A-DCS)	
IS-3267400	Unique Interface Specification for the Space Environment Monitor	
IS-2280354	Tiros-N Unique Interface Specification for the Digital Tape Recorder	
2629668	General Design and Test Requirements, NOAA-K, L, M	
PS-2285033	Specification for Tiros-N Electrical Aerospace Ground Equipment (NAGE)	
IS-2285780	Tiros-N Unique Interface Specification for the High Resolution Infrared Sounder/2 (HIRS/2)	
IS-23033278	ATN Unique Instrument Interface Specification for the Search and Rescue Repeater (SARR) for NOAAs -K, -L, & -M	
IS-3267401	Unique Interface Specification for the Search and Rescue Processor (SARP-2)	Mod 548 CCR 1949R1
IS-23033280	Unique Instrument Interface Specification for the Search and Rescue Processor (SARP-3)	
PS-2303078	Performance Specification Extended Power-On Processor Software (XPOPS)	
IS-2295548	Tiros-N Unique Instrument Interface Specification for the Solar Backscatter Ultraviolet Spectral Radiometer (SBUV/2)	
IS-2285557	Interface Specification Tiros-N Satellite Ground System	
2613246	Contamination Monitoring Plan, Witness Mirror, Test	
3267411	Reliability Program Plan for NOAA-K,L,M	

2308226	Configuration Management Plan for Tiros
IS-2624483	Unique Interface Specification for the Advanced Microwave Sounding Unit-A2 (AMSU-A2)
IS-2617547	Unique Instrument Interface Specification for the Advanced Microwave Sounding Unit Module-A1 (AMSU-A1)
IS-2613442	Unique Instrument Interface Specification for the Advanced Microwave Sounding Unit-B (AMSU-B)
IS-20046415	Unique Instrument Interface Specification for the Microwave Humidity Sounder
3267408	System Safety Implementation Plan (SSIP) for NOAA K,L,M Satellites
2295960	ATNAGE Requirements
PS-2285510	General Requirements Tiros-N Software
3267412	Quality Assurance Program Plan for NOAA-K,L,M
	Tiros Safehold System Requirements Document, Oct. 1995, ECN TRP 2904, CCR2413 CCR 2413 Mod 445
IS-20082617	Unique Instrument Interface Specification for the Solid State Recorder CCR 1594B Mod 426

3 REQUIREMENTS

3.1 SYSTEM DEFINITION

The satellite system shall sense, acquire, process, and relay meteorological and other Earth and space observation data.

3.1.1 General Description

The satellite comprises the integrated satellite system, interface with launch vehicle systems, flight computer programs, and support equipment.

3.1.1.1 Satellite

The satellite is defined as the total operational orbiting vehicle and shall consist of two major components: the spacecraft bus and the GFE payload. The satellite shall have two configurations: launch and preacquisition; and operational. These configurations are illustrated in Figures 1 and 2, respectively. The spacecraft bus shall be functionally divided into the following elements:

- Structure subsystems
- Thermal control subsystem
- Reaction control and propulsion subsystem
- Attitude determination and control subsystem
- Electrical power and distribution subsystem
- Communications subsystems
- Command and control subsystem
- Data-handling subsystem
- Payload adapter hardware

3.1.1.2 Computer Programs

The flight computer programs consist of the software for the onboard computers necessary to fully support the operational tests, launch, ascent, and in-orbit operations. This software shall include the required attitude determination and control, command and control, launch guidance, and interrupt driven controlling executive functions.

3.1.1.3 Support Equipment

The support equipment encompasses all hardware and software needed to make the satellite system operational in its intended environment when not directly engaged in the performance of its mission. The ground-support equipment (GSE) consists of the deliverable equipment including mechanical and electrical checkout, servicing, propellant and pressurant loading, simulations, and all software modules. The factory- support equipment consists of the nondeliverable checkout software and equipment.

3.1.1.4 Launch Vehicle Systems

The launch vehicle system consists of the launch vehicle, launch-site facilities, and support equipment as shown in Figure 3. On NOAA-K,L, and M the launch vehicle consists of the Titan-II booster and fairing. On NOAA N and N-prime the launch vehicle consists of the Delta II booster and fairing.

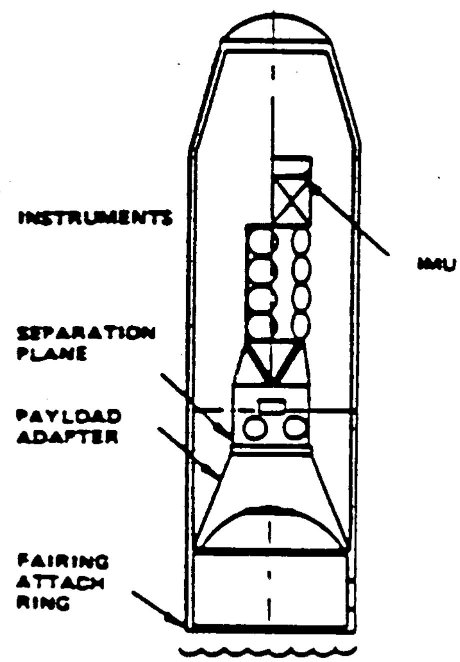

Figure 1. Launch Configuration

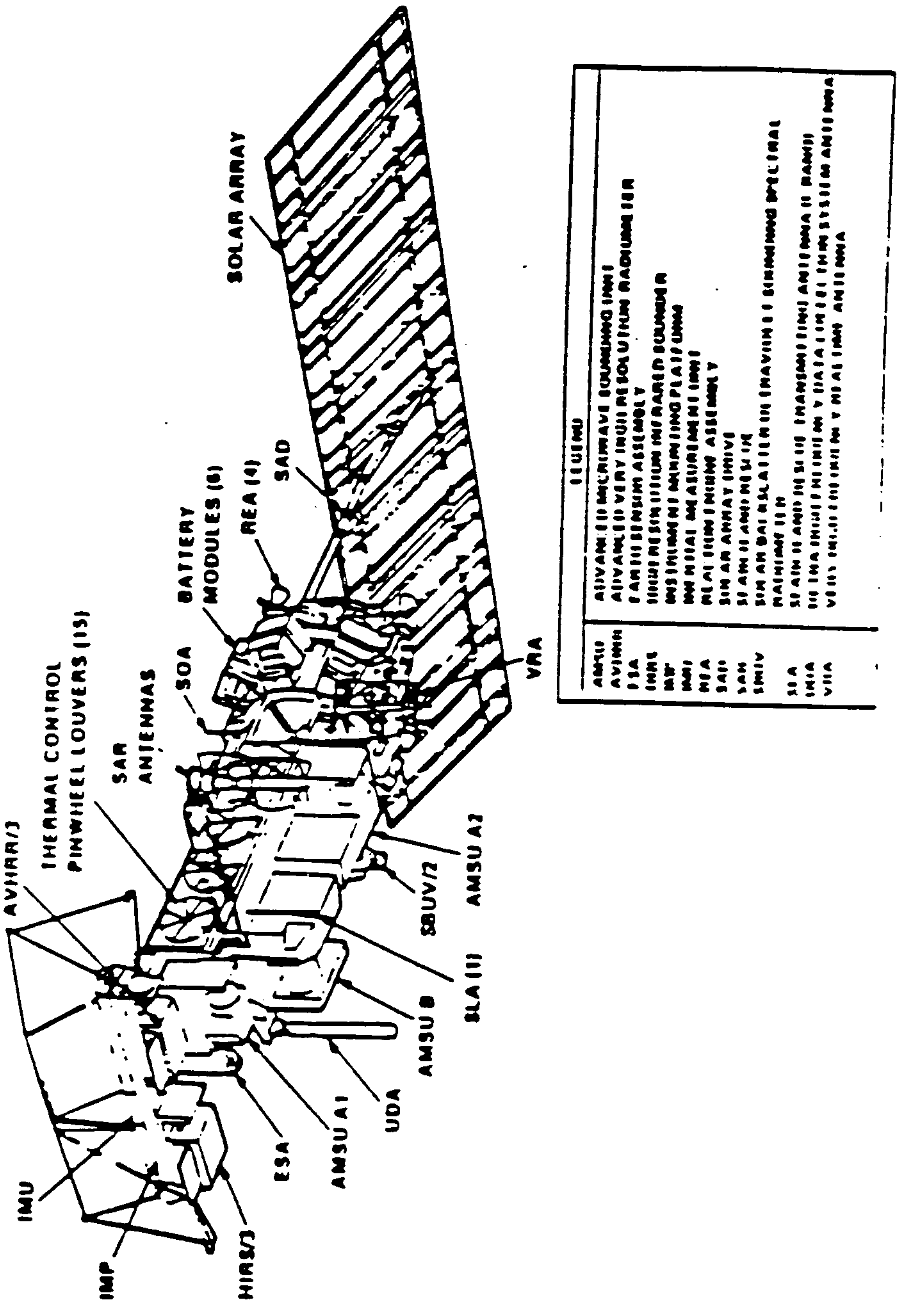

Figure 2. Typical Orbit Configuration Isometric (NOAA-K through -M)

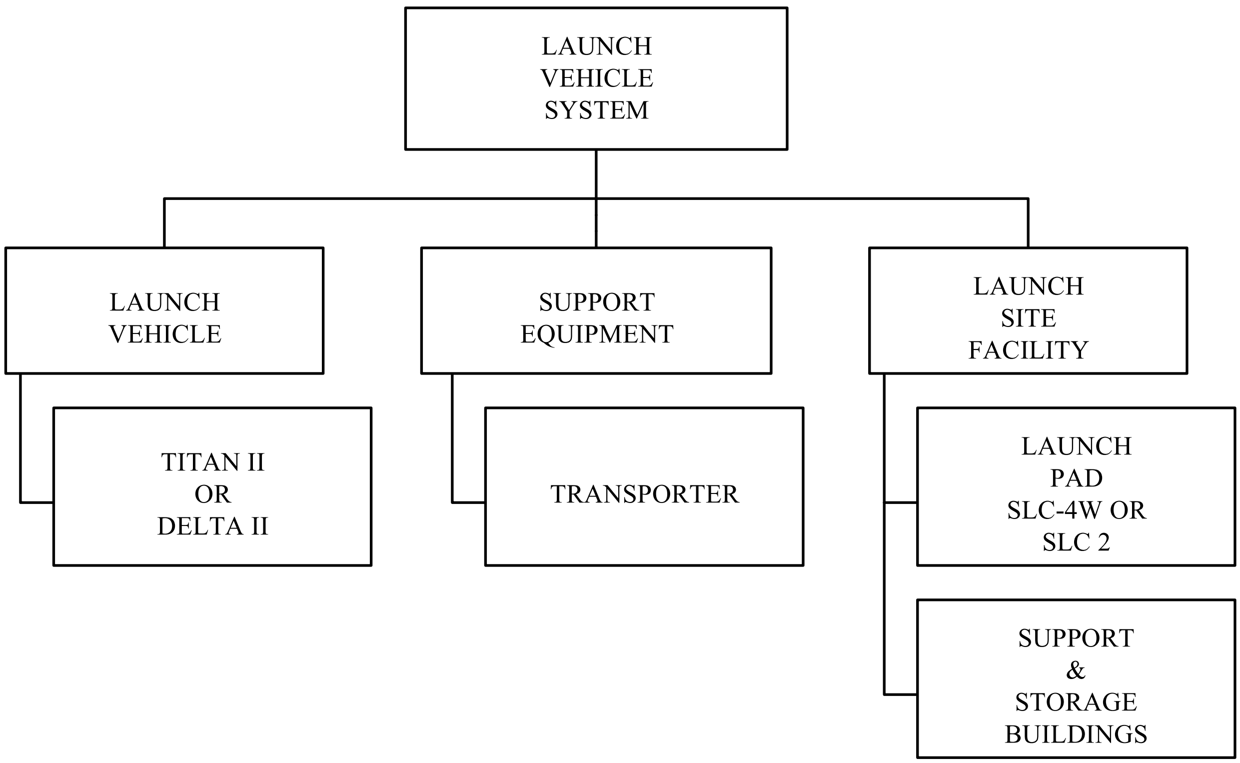

Figure 3. Launch Vehicle System Functional Areas

3.1.2 Mission

The satellite system shall support the mission by placing into orbit and supporting the instrument payload.

3.1.3 Satellite Functional Schematic

The satellite system functional schematic diagram shall be as shown in Figure 4. The interrelationships are briefly described in paragraph 3.1.6.

3.1.4 Interface Definition

The satellite shall be physically and functionally compatible with the interfacing elements shown in Figure 5. Detailed interface requirements shall be as documented in the interface specifications listed in Table 1.

3.1.4.1 Launch Vehicle System Interface Specification

Interface requirements for the NOAA-K, L, and M satellites and launch vehicle systems (noted in Figure 5) shall be specified in the Titan II ICD to the ATN spacecraft (NOAA-KLM), ICD-TII-25004.

Interface requirements for the NOAA-N satellite and launch vehicle systems (noted in Figure 5) shall be specified in the Delta II National Oceanic and Atmospheric Administration – N (NOAA-N) Mission Specification mdc 00h0072 October 2001 to the ATN spacecraft (NOAA-N). NOAA-N prime TBD. MOD 481 CCR 1891A

3.1.4.2 Instrument Interface Specification

General interfaces between the government-furnished equipment (GFE) instruments and the spacecraft that are common for all units shall be as documented in the "ATN-KLM General Instrument Interface Specification." Detailed unique interfaces for each instrument shall be as specified in the documents listed in Table 1.

3.1.4.3 Standard Recorders

Spacecraft interfaces with the digital data recorders shall be as specified in IS-2280354 (KLM) or UIS20082617) (MNN').

3.1.5 Government-Furnished Satellite Equipment List

Each satellite system shall be designed to incorporate the GFE listed in Table 2.

3.1.6 Operational and Organizational Concepts

The satellite system segment interrelationships with the program shall be as follows:

- The satellites will be launched from the United States Air Force (USAF) Space Launch Complex 4W (NOAA-K,L,M) or 2W (NOAA-N and N-prime) at Vandenberg Air Force Base (VAFB) with the launch vehicle contractor and the satellite contractor jointly conducting launch operations. Typical in-flight events commencing with liftoff and terminating with the achieving of final orbit will be as shown in Figures 6a (NOAA-K,L,M) and Figure 6b (NOAA-N and N-prime).

- After completion of the launch sequence, an in-orbit checkout phase will be conducted by NASA. Following successful completion of this checkout phase, the satellite will be turned over to NOAA for operational use.

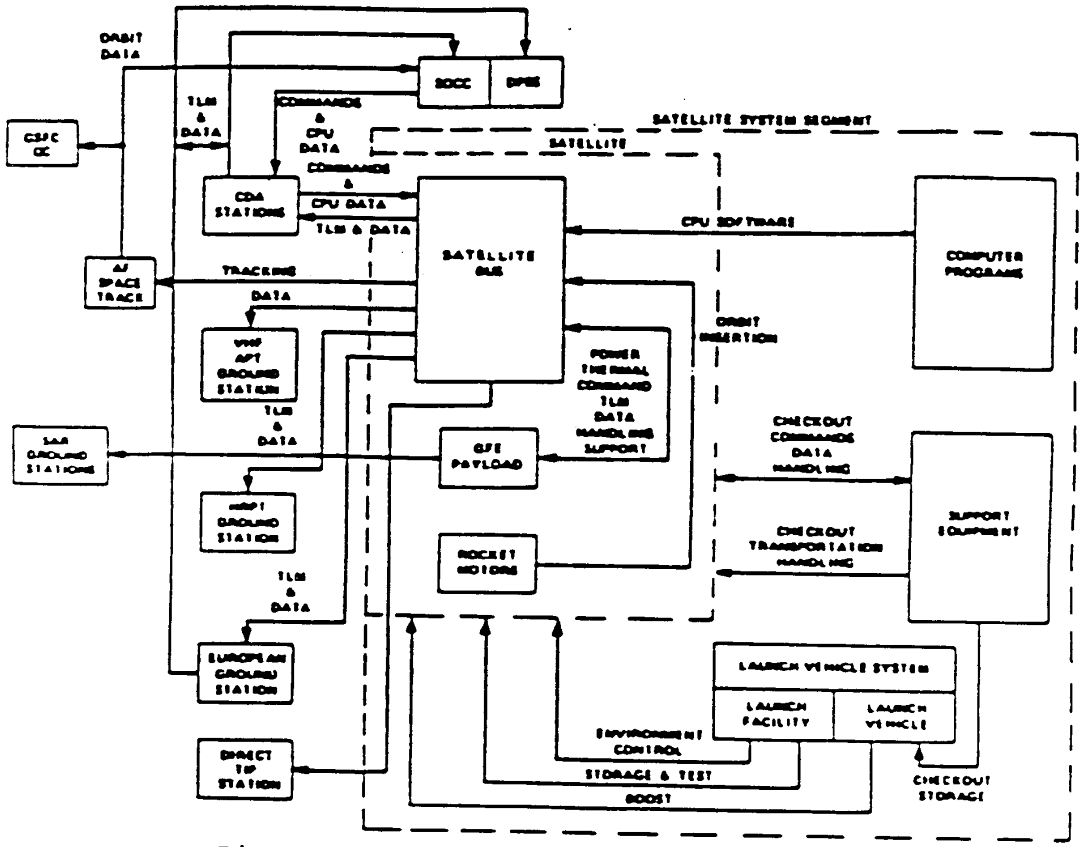

Figure 4. System Functional Schematic

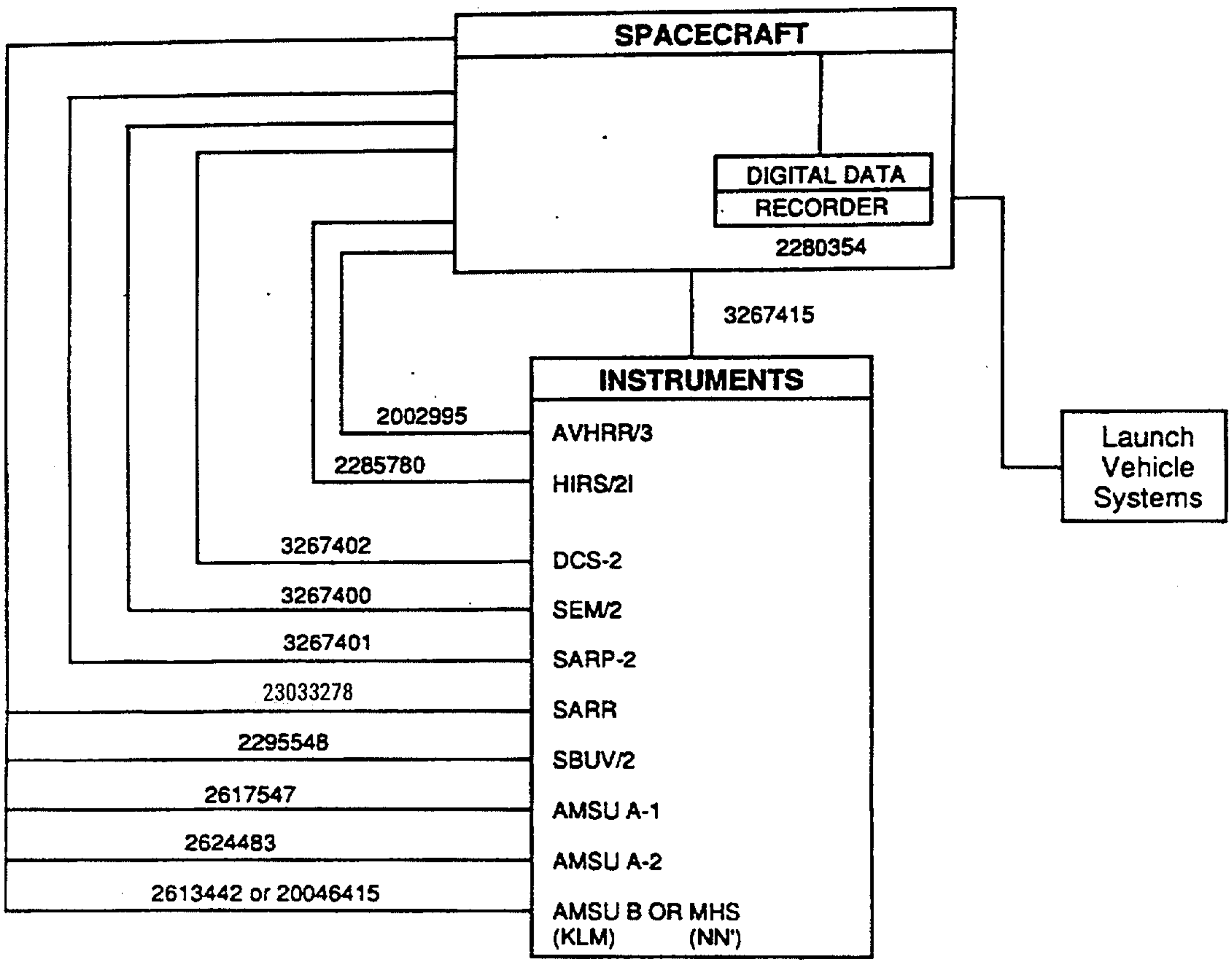

Figure 5. Satellite System Interfaces

Table 1 Interface Specifications

DESIGNATION	TITLE
ICD-TII-25004	Interface Control Document to the Advanced Tiros-N Spacecraft (NOAA-K, L, M)
IS-3267415	ATN-KLM General Instrument Interface Specification
IS-2002995	Unique Interface Specification for the Advanced Very High Resolution Radiometer (AVHRR-3)
IS-3267402	Unique Interface Specification for the Data Collection and Location Subsystem DCS-2
TBS	Unique Instrument Interface Specification for the Advanced Data Collection and Location System (A-DCS)
IS-3267400	Unique Interface Specification for the Space Environmental Monitor
IS-2280354	Tiros-N Unique Interface Specification for the Digital Tape Recorder (DTR)
UIS20082617	Tiros-N Unique Interface Specification for the Solid State Recorder (SSR)
IS-2285780	Tiros-N Unique Interface Specification for the High Resolution Infrared Sounder/2 (HIRS/2)
IS-23033278	ATN Unique Instrument Interface Specification for the Search and Rescue Re (SARR) for NOAAs -K, -L, & -M
IS-3267401	Unique Interface Specification for the Search and Rescue Processor (SARP-2)
TBS	Unique Instrument Interface Specification for the Search and Rescue Processor (SARP-3)
IS-2295548	Tiros-N Unique Instrument Interface Specification for the Solar Backscatter Ultraviolet Sounding Radiometer (SBUV/2)
IS-2624483	Unique Interface Specification for the Advanced Microwave Sounding Unit-A2 (AMSU-A2)
IS-2617547	Unique Instrument Interface Specification for the Advanced Microwave Sounding Unit A1 (AMSU-A1)
IS-2613442	Unique Instrument Interface Specification for the Advanced Microwave Sounding Unit-B (AMSU-B)
IS-20046415	Unique Instrument Interface Specification for the Microwave Humidity Sounder
MDC 00H0072	National Oceanic and Atmospheric Administration – N (NOAA-N) Mission Specification October 2001

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CCR 1891A

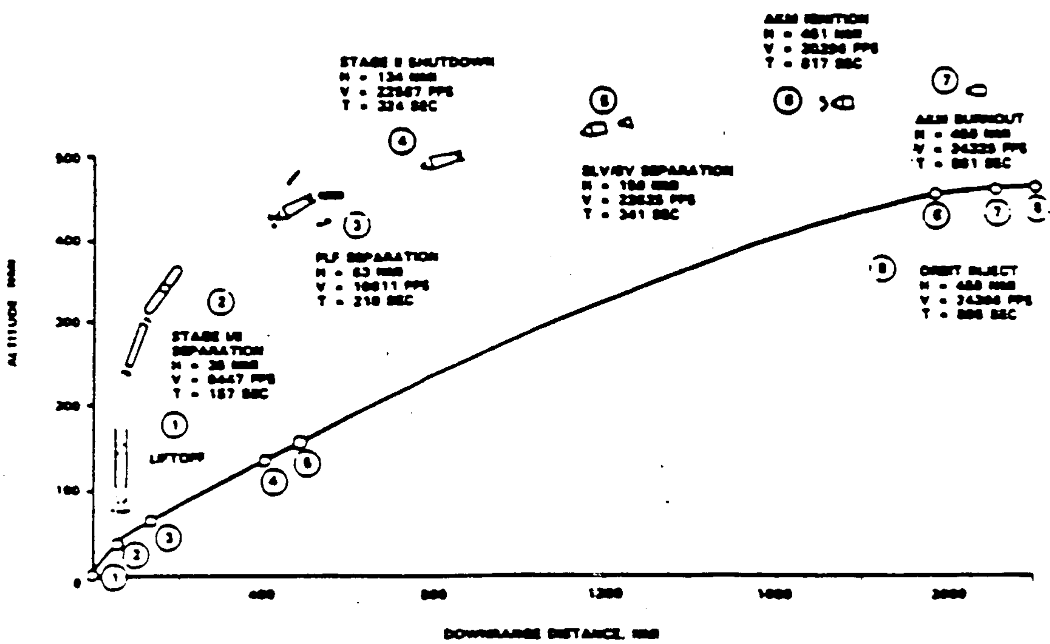
Table 2 Government-Furnished Satellite Equipment List

Item	K	L	M	N	N-prime
A-DCS/-3	-	-	-	-	X
SARP/-3	-	-	-	-	X
DTR (Note 2)	X	X	X	-	-
AVHRR/3	X	X	X	X	X
HIRS/3	X	X	X	X	X
DCS/-2	X	X	X	X	-
SEM/2 (TED, MEPED)	X	X	X	X	X
SARP/-2	X	X	X	X	-
SARR	X	X	X	X	X
SBUV/2(Note 1)	DUMMY	X	X	X	X
AMSU-A1	X	X	X	X	X
AMSU-A2	X	X	X	X	X
AMSU-B	X	X	X	-	-
MHS	-	-	-	X	X
SSR (Note 3)	-	-	X	X	X

Notes: 1. DUMMY denotes a physical simulation model with proper weight, thermal characteristics, and appropriate electrical terminations.

2. One digital tape recorder (DTR) consists of one electronics assembly and two tape transport assemblies. K & L each have five DTRs. M has four DTRs.

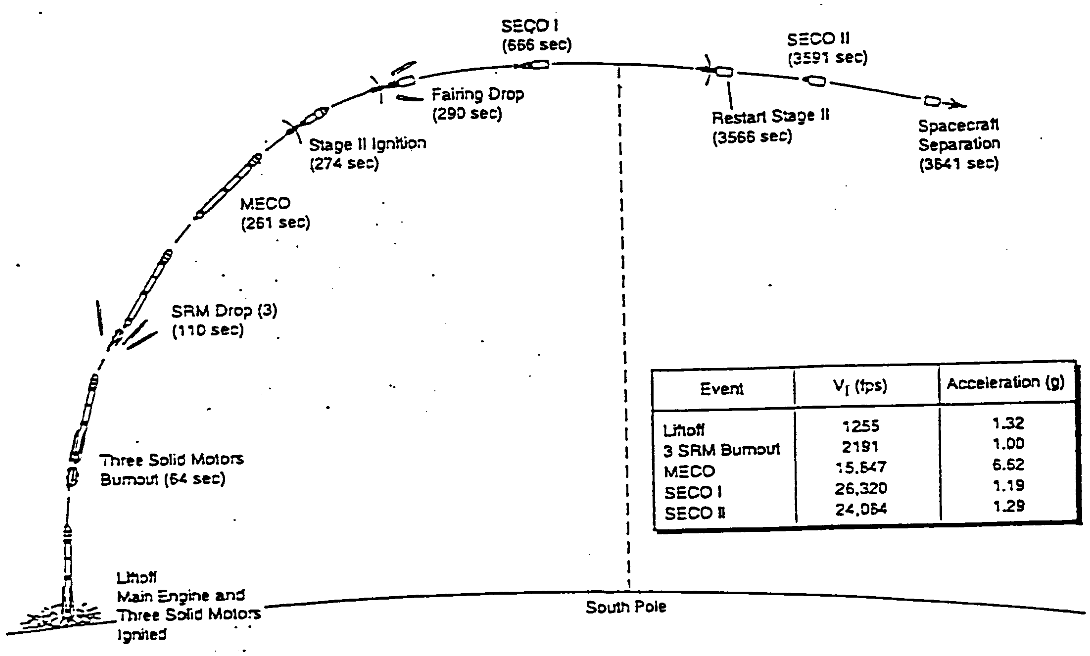
3. The M spacecraft has one Solid State Recorder (SSR). N&N' each have three Solid State Recorders (SSRs). The SSRs are also compatible with KLM. Each SSR replaces one or two DTRs.



SUN-SYNCHRONOUS MISSION SEQUENCE OF EVENTS (WITHOUT ACS)	
TIME (SEC)	EVENT
−5.0	GO INERTIAL
−2.0	STAGE I IGNITION SIGNAL
0.0	LIFTOFF
9.0	ROLL TO FLIGHT AZIMUTH
19.0	END STAGE I ROLL MANEUVER
20.0	START PITCH RATE 1
25.0	DRIVE ANGLE OF ATTACK TO ZERO (PITCH RATE 2)
30.0	BEGIN ZERO LIFT
110.0	END ZERO LIFT; START PITCH RATE 3
135.0	START PITCH RATE 4
156.9	STAGE I END STEADY STATE
157.7	STAGE I/II SEPARATION
161.0	BEGIN LINEAR SINE STEERING
210.1	PAYLOAD FAIRING SEPARATION
324.4	STAGE II SHUTDOWN
341.0	SV SEPARATION
816.8	AKM IGNITION
860.6	AKM BURNOUT
870.3	BEGIN SPACECRAFT VELOCITY TRIM
884.7	END SPACECRAFT VELOCITY TRIM

NOTE: THE EXACT SEQUENCE WILL BE AS SPECIFIED IN THE TITAN II TO ATN NOAA-K THROUGH -M ICD. THIS DATE IS FOR GENERAL INFORMATION ONLY AND WILL NOT BE UPDATED.

Figure 6a. Typical Launch Sequence Summary (NOAA-K, L, M)



Event	V_I (fps)	Acceleration (g)
Liftoff	1255	1.32
3 SRM Burnout	2191	1.00
MECO	15.647	6.52
SECO I	26,320	1.19
SECO II	24,064	1.29

NOTE: THE EXACT SEQUENCE WILL BE AS SPECIFIED IN THE DELTA II TO ATN N AND N-PRIME ICD. THIS DATA IS FOR GENERAL INFORMATION ONLY AND WILL NOT BE UPDATED

Figure 6b. Typical Launch Sequence Summary (NOAA-N and N-prime)

- The operational system will consist of two satellites to collect data for relay to the central processing facility and to remote users. The two satellites will be nominally positioned in mutually perpendicular Sun-synchronous orbital planes: a morning descending node orbit and an afternoon ascending node orbit with a nominal altitude difference of 20 nmi.

- The satellite will be commanded during its life by the Satellite Operations Control Center (SOCC) through the command and data acquisition (CDA) stations. The CDA stations will receive telemetry data. Data transmitted from the satellite will be received by the CDA stations, automatic picture transmission ground stations, high resolution picture transmission stations, and a European station, direct TIP transmission stations and SAR stations. The Air Force Space Tracking Ground Network will provide satellite tracking and orbit determination. The USAF Satellite Control Facility (SCF) will provide early orbit and emergency telemetry reception.

3.2 CHARACTERISTICS

The satellite shall be as physically and functionally identical as possible with the NOAA-J spacecraft. Differences shall be limited to those that are essential to the accomplishment of the requirements included or referenced within this specification, or are necessitated by design obsolescence.

3.2.1 Performance Characteristics

3.2.1.1 Satellite

a. Booster

NOAA-K,L,M

The launch vehicle shall be the U.S. Air Force Titan II. The major system elements and interfaces during launch to final orbit shall be as described in the Titan II ICD to the ATN spacecraft (NOAA-KLM), ICD-TII-25004.

The launch vehicle will be responsible for delivering the satellite to a targeted trajectory position for separation and takeover by the satellite ascent guidance system.

NOAA-N and N-prime

The launch vehicle shall be the Delta II (7320). The major system elements and interfaces during launch to final orbit shall be as described in the Delta II National Oceanic and Atmospheric Administration – N (NOAA-N) Mission Specification MDC 00H0072 to the ATN spacecraft (NOAA-N). NOAA-N prime TBD. MOD 481 CCR 1891A

The launch vehicle will be responsible for delivering the satellite to a targeted orbit that is biased from the mission orbit by the nominal satellite separation velocity.

b. Orbit

NOAA-K,L,M

Commencing with, and including, separation from the launch vehicle, the satellite shall perform all functions necessary to place it into a final nominally circular Sunsynchronous orbit of 450 or 470 nmi mean altitude. However, the satellite shall function operationally in orbit altitudes between 400 and 500 nmi. The satellite shall achieve the following final orbit within the stated 3σ tolerances contingent on proper performance of the launch vehicle.

- Sun-synchronous with the angle between the orbit normal and the solar vector (Sun angle) limited to a range of 0 to 80 degrees over a two-year operational life.

- NASA will select the time of nodal crossing for initial injection with a tolerance of ±5 minutes of time (maximum launch window - 10 minutes) so that the Sun angle will remain within the preceding solar vector range for an operational lifetime of 2 years assuming injection within the limits of the Titan II ICD to the ATN spacecraft (NOAA-KLM), ICD-TII-25004.

  The selected orbits are:

  1. Orbit #1--descending AM orbit with target parameters of inclination = 98.696 degrees and circular orbit radius = 3888.148 nmi

  2. Orbit #2--ascending PM orbit with target parameters of inclination = 98.856 degrees and circular orbit radius = 3908.148 nmi

  3. Orbit #3--ascending PM orbit with target parameters of inclination = 98.7442 degrees and circular orbit radius = 3908.148 nmi

  4. Orbit #4—descending AM orbit with target parameters of inclination = 98.7465 degrees and circular orbit radius = 3888.148 nmi

- Orbit altitude: 450 +10 nmi or 470 ±10 nmi (selected by NASA) above a spherical Earth radius of 3438.15 nmi (ascent guidance software constant).

- Orbit inclination tolerance: ±0.15 degree.

- Apogee to perigee distance difference: <30 nmi.

NOAA-N and N-prime

The launch vehicle will place the satellite into a nominally circular Sunsynchronous orbit of 450 or 470 nmi mean altitude. However, the satellite shall function operationally in orbit altitudes between 400 and 500 nmi. The satellite shall separate from the launch vehicle. After separation, the satellite shall be in the following final orbit within the stated 3σ tolerances contingent on proper performance of the launch vehicle.

MOD 551 CCR 1975

- Sun-synchronous with the angle between the orbit normal and the solar vector (Sun angle) limited to a range of 50 to 80 degrees over a two-year operational life.

- NASA will select the time of nodal crossing for initial injection with a tolerance of ±5 minutes of time (maximum launch window - 10 minutes) so that the Sun angle will remain within the preceding solar vector range for an operational lifetime of 2 years assuming injection within the limits of the Delta II National Oceanic and Atmospheric Administration – N (NOAA-N) Mission Specification MDC 00H0072 October 2001to the ATN spacecraft (NOAA-N). NOAA-N prime TBD. MOD 481 CCR 1891A

- Orbit altitude: 450 +10 nmi or 470 ±10 nmi (selected by NASA) above a spherical Earth radius of 3438.15 nmi (ascent guidance software constant).

- Orbit inclination tolerance: ±0.15 degree.

- Apogee to perigee distance difference: <30 nmi.

c. Orbital Operations -- For the specified operational life, the satellite shall:

- Be physically and functionally compatible with the GFE payload.
- Maintain Earth orientation.
- Provide electrical power sufficient to perform the mission for all required Sun angles.
- Collect, store, and transmit GFE payload data over the required links.
- Receive and transmit for verification, command and program data, execute valid storage and real-time commands, and control conditions under which program data shall be accepted.
- Collect, store, or transmit telemetry data over the required communications links.
- Maintain all satellite components, including GFE, within allowable, in-specification, operating temperature limits during component operations; and maintain temperatures at least within allowable, nonoperating limits at all other times.
- Specific requirements for these functions are detailed in paragraph 3.6 of this specification.

3.2.1.2 Support Equipment

The support equipment shall perform the functions required to: inspect, test, adjust, calibrate, measure, assemble, disassemble, handle, transport, safeguard, store, actuate, service, and maintain the satellite during all factory-to-launch operations; and check out and launch the satellite at the launch site and assist in crew training and validation of NOAA/SOCC readiness to support post launch activities. As a minimum, these functions shall include the following:

- Balance the satellite in its assembled configuration.
- Monitor and evaluate performance of the satellite and its components and subsystems during all phases of testing.
- Evaluate the end-to-end performance during functional, environmental, and integrated system development, qualification, and acceptance tests of the satellite and its subsystems.
- Verify the static mechanical interfaces.
- Simulate those signals during functional and integrated system tests that would normally be transmitted from the ground to the satellite.
- Perform end-to-end testing of the integrated satellite in its launch configuration during system checkout testing.
- Perform servicing, environmental conditioning, electrical power; electrical grounding, and mechanical support for satellites during all ground operations.
- Perform loading and detanking of propellants and pressurants for the satellite

3.2.2 Physical Characteristics

3.2.2.1 Satellite

a. Mass Properties

- Weight -- The weight of the NOAA-K,L, and M satellites at launch shall be 4,953 $\pm$ 21 pounds. The weight of the NOAA-N and N-prime satellites at launch shall be 3200 pounds (tbr). The nominal weight distribution by subsystem shall be as given in Table 3a for NOAA-K,L, and M and in Table 3b for NOAA-N and N-prime. The spacecraft margin weight indicated in the table shall be for apportionment between subsystems as design

Table 3a Nominal Weight Distribution NOAA-K, L, and M (pounds)

Assembly	Estimated Weight
Structure	642.1
Thermal	77.3
ADACS (dry)	117.6
Power	527.0
Communications	58.7
Command and Control	68.9
Data Handling	39.9
GFE	982.5
Harness	233.0
AKM Case	145.6
Reaction Control	68.0
Balance Weight	38.1
Subtotal	2998.7
Margins/Ballast	157.8
Adapter	122
S/C Dry Weight	3251.2
N_2H_4	62.8
GN_2	10.4
AKM Expendables	1595.6
S/C at Liftoff	4947.3

Table 3b Nominal Weight Distribution NOAA-N and N-Prime (pounds)

Assembly	Estimated Weight
Structure	642.1
Thermal	77.3
ADACS (dry)	117.6
Power	527.0
Communications	58.7
Command and Control	68.9
Data Handling	39.9
GFE	982.5
Harness	233.0
Reaction Control	68.0
Balance Weight	38.1
Subtotal	2853.1
Margins/Ballast	123.7
Adapter	150.0
S/C Dry Weight	3126.8
N_2H_4	62.8
GN_2	10.4
S/C at Liftoff	3200.0

requirements dictate. The GFE payload margin weight allowance shall be maintained for changes in, or additions to, instrument payload and any resulting increase in required support equipment including balance weight.

- Center of Mass -- The coordinates and tolerances of the satellite center of mass shall be compatible with launch vehicle interface requirements.

b. Dynamic Envelope

NOAA-K, L, and M

The satellite, in launch configuration, shall remain within the dynamic envelope of the fairing and mate with the launch vehicle as specified in the Titan II ICD to the ATN spacecraft (NOAA-KLM), ICD-TII-25004.

NOAA-N and N-prime

The satellite, in launch configuration, shall remain within the dynamic envelope of the fairing and mate with the launch vehicle as specified in the Delta II National Oceanic and Atmospheric Administration – N (NOAA-N) Mission Specification MDC 00H0072 October 2001 to the ATN spacecraft (NOAA-N). NOAA-N prime TBD.

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c. Transport and Storage Life -- The flight configured satellite shall be suitably packaged for transport in a manner that will prevent damage. The satellite shall be shipped in a container whose environment shall be monitored for all controlled conditions; shock, temperature, and humidity. The storage life of the satellite when stored in a controlled environment shall be at least 3 years without degradation to required in-orbit operational longevity.

d. Security Criteria -- The command uplink shall be encrypted and authenticated to minimize the probability of unauthorized commanding.

3.2.3 Reliability

The satellite, excluding GFE payload, shall be designed to have a minimum 2-year orbital lifetime. This includes ascent, acquisition, and on-orbit operations.

3.2.4 Maintainability

The integrated spacecraft design shall be such that component failures during ground test, checkout, and storage may be repaired without degrading the performance or reliability characteristics of other components.

3.2.5 Environmental Conditions

3.2.5.1 Satellite

a. Transportation, Handling, and Storage Environment -- The satellite shall be capable of operating within specification limits after exposure to both controlled and natural environments while in nonoperating condition during transportation, handling, and storage. These environments include those experienced during fabrication, storage for at least 3 years, handling, transportation, and erection at the launch site. Controlled environments shall be provided when necessary to bring the experienced temperature, humidity, shock, and vibration to levels less severe than those pertaining to launch, ascent, and orbital operations. The satellite shall be protected from air transportation temperatures of -40° to +66°C with an 18-degree variation per minute for 5 minutes and truck transportation temperatures of -40° to +66°C with a 3-degree variation per minute. The maximum range in altitude will occur during air shipment from sea level to 15,000 meters at a maximum vertical velocity of 30 meters/second.

b. Prelaunch Environment -- The contractor shall assess the launch site of the satellite to determine that the environ-ment meets satellite requirements for launch, ascent, and orbital operations. The contractor will advise NASA of the results of this assessment.

c. Flight Environment -- The satellite shall be capable of withstanding or shall be protected against the following flight-induced environments without performance degradation.

- Temperature
  - Radiation and convection heating from heat shield prior to heat shield jettison
  - Aerodynamic heating after heat shield jettison
  - Solar radiation
  - Emitted and reflected thermal radiation from the Earth
  - Heating from operational equipment
  - Radiation to space

- Altitude -- Sea level to vacuum conditions of the orbital altitude

- Acoustic Field -- The acoustic criterion will be as specified in Table 4a for NOAA-K,L,M, and in Table 4b for NOAA-N and N-prime..

- Sinusoidal Vibration -- A sinusoidal test shall be conducted as described in section 4.4.4.

- Structural Loads

NOAA-K,L,M

The satellite structural loads shall be determined by test and analysis. An analytical model shall be generated and test verified (i.e., model survey, low level sine sweep, etc.) to 100 Hz to verify that the spacecraft dynamic characteristics such as frequency and mode shapes have been adequately represented. A coupled loads analysis shall be conducted to determine the limit (flight) loads. The steady state accelerations are identified as follows:

Event/Axis	Acceleration, g, g/in
Liftoff	
Axial	-0.5 to +3.0
Lateral	±3.0
Maximum Airloads	
Axial	+1.0 to +3.0
Lateral	±2.5
Stage I Shutdown (Depletion)	
Axial	-3.0 to +9.0
Lateral	±3.0
Stage II Shutdown (Command)	
Axial	-4.0 to +12.0
Lateral	±2.0

Table 4a Tiros Acoustic Environment (NOAA-K,L,M)

(MDSSC 10' x 20' UPLF)

(Ref: dB = 0.00002 Pa)

1/3 Octave Band Center Frequency (Hz)	Predicted Free Field (Empty) W/1.5" Blankets (dB)	KLM Design Criteria (dB)	KLM Test Criteria (Flight Levels) (dB)
50.0	112.2	119.0	119.7
63.0	113.7	121.5	121.4
80.0	117.3	124.0	125.0
100.0	117.9	122.0	125.6
125.0	119.1	125.0	126.8
160.0	118.4	124.5	126.1
200.0	118.1	125.0	125.8
250.0	119.0	125.5	126.7
315.0	120.0	124.5	127.7
400.0	119.9	123.5	127.6
500.0	120.0	123.0	127.7
630.0	119.9	121.5	127.6
800.0	118.8	119.0	126.5
1000.0	115.5	115.0	123.2
1250.0	115.8	112.5	123.5
1600.0	113.5	109.0	121.2
2000.0	111.5	106.5	119.2
2500.0	110.0	106.5	117.7
3150.0	108.5	105.5	116.2
4000.0	104.6	102.5	112.3
5000.0	102.7	103.0	110.4
6300.0	100.1	102.5	107.8
8000.0	96.0	101.5	103.7
10000.0	93.5	96.5	101.2
OASPL (dB)	130.3	134.5	138.0

Note 1: Test Tolerances:	50-2000 Hz	±3 dB	
	2000-10000 Hz	+3, -6 dB	(Higher tolerances are permitted due to facility capability above 5000 Hz)
	OASPL	±1 dB	

Table 4b Predicted PLF Acoustic Environment for NOAA-N and N-prime

(Ref: dB = 0.00002 Pa)

1/3 Octave Band Center Frequency (Hz)	Free Field (Empty) W/1.5" Blankets (dB)
31.5	117.5
40.0	120.5
50.0	123.5
63.0	126.0
80.0	128.0
100.0	129.0
125.0	129.5
160.0	130.0
200.0	130.0
250.0	130.0
315.0	130.0
400.0	130.0
500.0	128.5
630.0	125.5
800.0	122.5
1000.0	120.0
1250.0	117.5
1600.0	116.0
2000.0	115.0
2500.0	114.5
3150.0	113.5
4000.0	112.5
5000.0	110.5
6300.0	106.5
8000.0	102.5
10000.0	99.0
OASPL (dB)	139.9

Note: Payload weight assumed to be in 3,000 to 5,000 lb range. Accelerations include both steady state and dynamic.

The longitudinal and lateral accelerations are applied simultaneously. A static load test shall be conducted to 1.25 times the limit loads determined from the coupled loads analysis.

NOAA-N and N-prime

The satellite structural loads shall be determined by test and analysis. An analytical model shall be generated and test verified (i.e., model survey, low level sine sweep, etc.) to 100 Hz to verify that the spacecraft dynamic characteristics such as frequency and mode shapes have been adequately represented. A coupled loads analysis shall be conducted to determine the limit (flight) loads. The steady state accelerations are identified as follows:

Event/Axis	Acceleration, g, g/in
Liftoff/Transonic	
Axial	-0.2 to +2.5
Lateral	±2.5
MECO	
Axial	+7.4 static, ±0.6 Dynamic
Lateral	-0.1 to +0.1

Note: Payload weight assumed to be in 3,000 to 3625 lb range. Accelerations include both steady state and dynamic.

The longitudinal and lateral accelerations are applied simultaneously. A static load test shall be conducted to 1.25 times the limit loads determined from the coupled loads analysis.

- Radiation Environment -- The satellite shall be designed to withstand the energetic particle radiation specified in Metsat Charged Particle Environment Study, November 1985.

- Shock -- The satellite shall be designed to withstand the shock generated by the release of the separation band and deployable appendages.

- Radiated Emission -- For units that are new or significantly modified from those used on NOAA-J, the design and test requirements of paragraph 3.6.1.4.2 of IS-3267415 (ATN-KLM General Instrument Interface Specification) shall be applicable.

d. Contamination Control Requirements -- The spacecraft assembly and integration and test (I&T) operations shall be performed in Federal Standard FED-STD-209B Class 100,000 or better environment. Lower level assemblies shall be cleaned prior to installation in the spacecraft. The contractor shall monitor the molecular contaminants in the spacecraft assembly area. The contractor shall keep contaminant materials outside the spacecraft assembly area as required by E 6.12, "Controlled Clean Areas" and 2280728, "Process Standard Operating Instruction Controlled Clean Area." If contaminants exceed 50 micrograms/sq in/month, NASA may direct a spacecraft cleaning in accordance with existing procedures (2280887) and the contractor shall take appropriate action to prevent the reoccurrence of similar contamination caused by contractor actions.

3.2.5.2 Ground-Support Equipment

The GSE shall be capable of operating within specification limits during and after exposure to both controlled and natural environments experienced during test, prelaunch, and launch operations. The ground-support equipment shall be made capable of operating within specification after exposure to both controlled and natural environments

experienced during transportation, handling, and storage.

3.3 DESIGN AND CONSTRUCTION

3.3.1 Parts, Materials, and Processes

The satellite parts, materials, and processes shall be selected, designed, manufactured, and controlled in accordance with the requirements specified in S-480-26.1.

3.3.2 Nameplates and Product Marking

All spacecraft contract supplied equipment shall be permanently marked using characters 0.12-inch high with white epoxy marking paint on dark surfaces and black epoxy marking paint on light surfaces. Markings shall include:

- Equipment nomenclature
- Part number
- Serial number
- Contract number (or purchase order number)
- Manufacturer's name or trademark
- Manufacturer's part number (for non-MMAS equipment)

3.3.3 Workmanship

Workmanship on all components and parts shall meet the requirements specified in S-480-26.1. All personnel employed in critical processes shall be qualified and certified through a formal training program in accordance with either accepted industry practice or government agency practice.

3.3.4 Interchangeability

All equipment with the same identifying assembly shall be mechanically and electrically interchangeable with all other assemblies so identified.

3.3.5 Electrical and Electronic Safety Design Requirements

3.3.5.1 Spacecraft

a. As a goal, elements of redundant systems shall not pass through the same connector used by elements of the primary system.

b. Sufficient working space shall be provided around connectors for engaging and disengaging, particularly where wrenches are required.

c. Satellite electrical equipment shall be designed so that it is not possible to ignite or contribute to the ignition of adjacent materials regardless of the atmosphere.

d. Satellite system design shall incorporate features to protect personnel operating or checking the complete satellite from accidental contact where voltages exist in excess of 30 volts rms or 30 volts dc.

e. The external surfaces of the satellite shall be designed and constructed to ensure that all surfaces are always at dc ground potential. Thermal blankets need not be grounded unless the proper performance of instruments or antennas requires blanket grounding.

f. Where personnel access is likely to occur during system operations, barriers or guards shall be provided to prevent personnel from accidentally contacting bus bars, terminals, or other devices that may cause dangerous electric shock or short circuits.

g. As a goal, design layout shall be used to positively prevent inadvertent reversing or mismatching of electronic connectors.

3.3.5.2 Support Equipment

a. A means shall be designed or provided to positively prevent inadvertent reversing or mismating of electrical connections.

b. Design shall incorporate features to protect personnel operating or checking a complete unit from accidental contact where potential voltages exist in excess of 30 volts dc or 30 volts rms.

c. Means shall be provided to cut off power while units, assemblies, or portions thereof are being installed, replaced, or interchanged.

d. Personnel shall be protected from capacitor discharges when changing fuses, tubes, or other components where potential contact is possible.

e. The main power switch located on the equipment (clearly labeled as such) shall cut off all power to the equipment.

f. All external parts, surfaces, and shields shall be at dc ground potential at all times. An external or interconnecting cable, where ground is part of the circuit, shall carry a ground wire in the cable terminated at both ends in the same manner as the other conductors. Thermal blankets need not be grounded unless the proper performance of instruments or antennas requires blanket grounding;

g. The cable shield shall not be depended on for a current carrying ground connection (except for coaxial cables).

h. Antenna and transmission line terminals shall be at dc ground potential.

i. Plugs and convenience outlets for use with portable tools and checkout equipment shall have provisions for automatically grounding the frame or case of the tools when the plug is mated with the receptacle. Polarity of the outlet shall be verified correct before connecting to the system.

j. Ground connections to shields and other mechanical parts, except the chassis or frame, shall not be made to complete electrical circuits.

k. The path to ground from the equipment shall:

- Be continuous and permanent.
- Have ample carrying capacity to safely conduct any currents that it is possible to impose.
- Have impedance sufficiently low to limit the potential above ground and to facilitate the operation of the over current devices in the circuit. As a goal, inactive wires installed in long lines (conduit or cables) shall be grounded.
- Be designed to minimize the possibility of inadvertent ground disconnection.

1. All contacts, terminals, and similar devices that may cause dangerous electric shock or short circuits shall have barriers or guards installed to prevent accidental contact by personnel, tools, or conductive debris. The barrier or guard shall contain a warning to indicate the highest possible voltage potential.

m. Where design considerations require incompatible plugs and receptacles with similar configuration close by,

connectors shall be keyed and coded so that it will be impossible to insert the wrong plug in a receptacle or other mating unit.

n. Externally applied power shall be controlled by a power on-off switch located on the unit operating panel. An indicator lamp shall be provided to indicate power on the unit. The switch shall be configured so that a positive indication of power condition is provided during a power-off phase. The ground power supply shall contain the necessary internal design to prevent over voltage in excess of the spacecraft design limit and shall also limit the spacecraft input current in the event of a short circuit in the spacecraft or connecting harness.

o. Neither side of the supply voltage shall be connected directly to the chassis.

p. Design of mechanical equipment or portions thereof shall be such that maximum convenience and safety are provided to personnel during installation, operation, and maintenance. Suitable protection shall be provided to prevent contact with moving parts such as gears, fans, and belts when the equipment is operating. Sharp projections on cabinets, doors, and similar parts shall be avoided.

q. Equipment drawers or racks shall be designed to prevent accidental mechanical disengagement from slides when pulled out to the limit.

r. Power switches shall be located so that accidental contact by personnel will not place equipment in operation. Critical switches that can produce or induce hazardous conditions if inadvertently actuated will be guarded, shielded, or otherwise protected.

s. Where it is possible for ground electrical equipment to be exposed to explosive vapors in excess of 25 percent of their lower explosive limit while operating, the design shall ensure that normal operation of electrical and electronic equipment will not cause ignition or the associated equipment and its use will be approved by the required safety officer in accordance with WSMCR 127-1, Range Safety Manual, Vol. 1.

t. Electrical grounding connection capability shall be specified for all trucks and dollies. Electrical wiring shall be closed in chafe-resistant protective sheathing, and clamping shall stay clear of sharp edges and moving parts.

3.3.6 Handling and Transportation Safety Requirements

3.3.6.1 General

a. Spacecraft handling and transport operations are considered to be safety-critical operations and will conform to approved procedures.

b. All hoisting cranes interfacing with the spacecraft shall have evidence of being proof-load tested within 48 hours of use. All slings shall be tested yearly.

c. When a forklift is used for movement operations, the load shall be secured to the forks so that it cannot shift and topple.

d. Any equipment that in normal operation exposes personnel to surface temperatures of more than 60°C (140°F) as a result of inadvertent contact, or 49°C (120°F) during handling, shall be guarded.

e. The operation of switches or controls that initiate hazardous operations (e.g., ignition and movement of a crane) shall require the prior operation of a related or locking control. When practicable, the critical position of such a control shall activate a visual and auditory warning device in the affected work area.

f. Units shall be so located and mounted that access to them can be achieved without danger to personnel from electrical charge, heat, moving parts, chemical contamination, radiation, or other sources.

g. When access areas must be located over dangerous mechanical or electrical components, the access door or cover shall be designed to actuate an internal light when opened, and a highly visible warning label shall be provided on the outside of the door or cover.

3.3.6.2 Platforms

a. Self-locking or other fail-safe devices shall be incorporated on elevating stands and work platforms to prevent accidental or inadvertent collapsing.

b. Screen or safety mesh shall be installed on the underside of open gratings, platforms, or flooring surfaces where there is a possibility that small tools or parts may fall through the grating on workers or equipment beneath the platforms.

c. Plastic shields or similar materials shall be installed on the underside of perforated platforms, which extend over the satellite, to prevent contaminating particles from falling on the satellite.

3.3.6.3 Lifting Devices

a. Maximum safe working load, proof load, certification, and effective date shall be stenciled or tagged on lifting devices.

b. Handling and lifting equipment shall be designed not to cause any spacecraft contamination.

c. Handling and lifting equipment shall be designed with safeguards at the attach points to prevent damage to the spacecraft from handling.

d. Lifting devices shall be designed with rings, links, hooks, or eyes so that they can be safely and independently suspended on the safety hooks of hoisting devices.

3.3.6.4 Slings

Sling cables shall be sufficiently long to ensure that the angle formed by the cables at the hoist attaching point is 45 degrees maximum.

3.3.6.5 Hooks

Hooks shall be fitted with safety latches or other safety devices and will be shaped to prevent loads from slipping off.

3.3.6.6 Cradles and Jacks

a. Cradling or support devices and tiedowns conforming to the shape, size, weight, and contour of the load shall be specified.

b. Jacks shall have adjustable mechanical stops to ensure even lifting when several jacks are used.

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3.4 SECTION RESERVED

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3.5 SECTION RESERVED

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3.6 FUNCTIONAL AREA CHARACTERISTICS

3.6.1 Space Vehicle Characteristics

3.6.1.1 Structure Subsystem

The structure subsystem shall provide the mechanical framework to support the remaining spacecraft subsystems and the instruments throughout the launch and orbital phases of the mission. The structure of the satellite shall include:

- An instrument mounting platform (IMP) designed primarily to support instruments and attitude control sensors having field-of-view requirements
- An equipment-support module (ESM) designed primarily to support the remainder of the spacecraft electronic equipment
- A reaction control equipment (RCE) support structure (RSS) designed primarily to support the reaction control equipment, and on NOAA-K, L, and M to also support the propulsion equipment
- A solar-array support (SAS) designed to support the solar array panels and the array-drive mechanism
- Hardware and mechanisms required for all deployments and stage separations
- Unique payload adapter to provide mechanical interface between the satellite and the launch vehicle

The structure subsystem shall be configured as shown in Figure 7, and the coordinate system is as described in paragraph 3.6.1.li.

a. Weight and Volume

NOAA-K, L, and M

The spacecraft structure shall be designed to accommodate the total satellite mass and size, including the rocket motor and GFE payload within the size and volume limitations imposed by the heat-shield described in the Titan II ICD to the ATN spacecraft (NOAA-KLM) (ICD-TII-25004).

NOAA N and N-prime

The spacecraft structure shall be designed to accommodate the total satellite mass and size, including the GFE payload within the size and volume limitations imposed by the heat-shield described in the Delta II National Oceanic and Atmospheric Administration – N Mission Specification MDC 00H0072 October 2001 to the ATN spacecraft (NOAA-N). NOAA-N prime TBD.

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b. Booster Interface

NOAA-K, L, and M

The structure subsystem shall be capable of interfacing with the launch vehicle through the launch vehicle adapter. The spacecraft shall be separated from the booster by a hydrazine system. The separation system shall be designed to separate the spacecraft from the Titan II booster to preclude contamination of the spacecraft by the hydrazine (N_2H_4) and to provide a 3σ (99.736 percent) probability of not sustaining a post separation collision between spacecraft and booster.

The booster/spacecraft separation scheme shall be as specified in the Titan II ICD to the ATN spacecraft (NOAA-KLM)(ICD-TII-25004). The separation sequence shall utilize the IMU accelerometer and/or a Titan II vehicle discrete and/or backup timer.

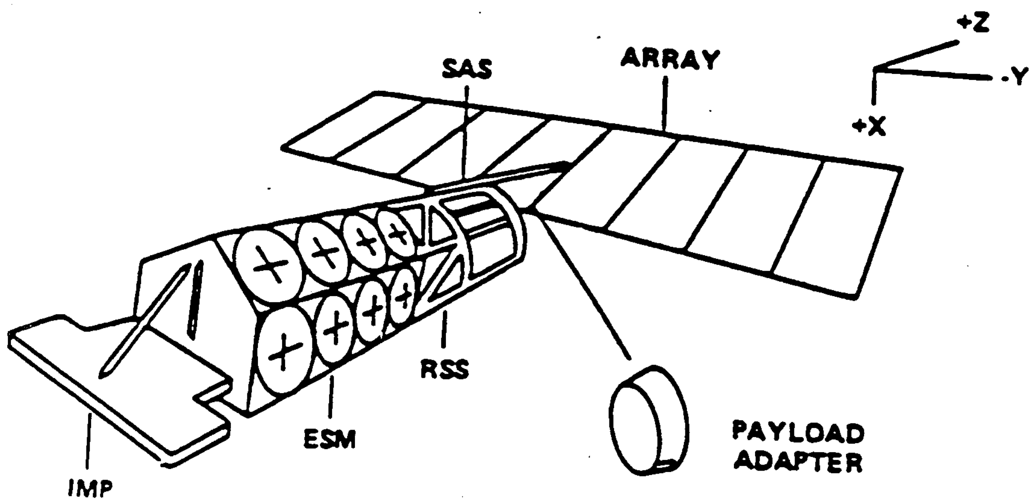

Figure 7. Typical Advanced Tiros Structure Subsystem

NOAA-N and N-prime

The structure subsystem shall be capable of interfacing with the launch vehicle through the launch vehicle adapter. The spacecraft shall be separated from the booster by springs. The separation system shall be designed to separate the spacecraft from the Delta II and to provide a 3σ (99.736 percent) probability of not sustaining a post separation collision between spacecraft and booster.

The booster/spacecraft separation scheme shall be as specified in the Delta II National Oceanic and Atmospheric Administration – N Mission Specification MDC 00H0072 October 2001 to the ATN spacecraft (NOAA-N). The separation sequence shall utilize the IMU accelerometer and/or a Delta II vehicle discrete and/or backup timer. NOAA-N prime TBD. MOD 481 CCR 1891A

c. Solar-Array Support -- The structure subsystem shall provide for mounting and deploying the solar array. This solar-array support shall be capable of completely deploying the array within 10 minutes after the deployment command. Hinges shall use positive detent mechanisms to lock the array in the deployed position. The solar-array support shall be designed and mounted on the RSS so that interactions of array/boom flexing and tracking with the attitude control system do not result in out of performance operation.

d. Equipment Support -- Provision shall be made for mounting all equipment necessary to fulfill the mission.

e. Thermal Layout -- The structure subsystem shall provide an arrangement of satellite components so that the thermal requirements of paragraph 3.6.1.2 are met. Thermal gradients between components requiring precise mutual alignment shall be minimized.

f. Electrical Layout -- Whenever possible, the structure subsystem shall allow components to be grouped functionally so as to shorten harness lengths. Components that generate electromagnetic interference (EMI) shall be placed as far as possible from EMI sensitive components.

g. Fields of View -- The structure subsystem shall allow for component arrangement so that the field of view of any instrument sensor, attitude determination sensor, or antenna is not obstructed throughout the operational life of the satellite. Motion of the solar array caused by tracking and flexing shall be considered.

h. Access -- The spacecraft structure shall allow access to all components, either through open areas, removable panels, or doors without jeopardizing the structural integrity of the remaining structure.

i. Satellite Coordinate System -- A right-hand, body-fixed coordinate system shall be used for references. The Z-axis shall be normal to the separation plane and pass through the center of the separation ring. The positive sense of the Z-axis shall be in the direction traveled, moving from the IMP to the RSS. The X-axis shall be normal to the Earth-facing side of the spacecraft with the positive sense being directed toward the Earth. The Y-axis shall be normal to the X and Z axes to form a complete right-hand coordinate system. The nominal instantaneous velocity vector shall be along the Y-axis. The origin of the system shall be located on the plane of separation between the spacecraft and the booster.

j. Alignment -- The structure subsystem shall provide for mounting satellite equipment so that critical alignment requirements are not exceeded. The criticality of alignment will depend on the particular sensor; instrument payload requirements shall be as referenced in paragraph 3.1.4.2. The spacecraft alignment reference shall be the Earth Sensor Assembly. The AMSU-A1, AMSU-A2, AMSU-B, HIRS/3, and AVHRR/3 shall be aligned on the spacecraft to each other to within 0.55 degrees.

- Instrument Mounting Platform -- The instrument mounting platform shall be fabricated and attached to the equipment support module so as to allow the required alignments to be met.

- Antenna Alignment -- Spacecraft antennas shall be mounted so that the reference axes are aligned to within 1 degree of nominal in the orbit configuration except for those antennas whose pattern versus angle are specified in the following paragraphs. Alignment, which is achieved through mechanical design tolerancing, is not a measured parameter.

k. Structural Design Requirements: Materials, Properties, and Allowables -- The design of the payload structure and its components shall provide satisfactory performance of its required mission in all phases of flight, launching, handling, stowage, transportation, and ground testing. All structural members of the spacecraft shall be designed to meet the following criteria:

- For single load path structures, the minimum guaranteed values ("A" values in MIL-HDBK-5) shall be used. For multiple load path structures, the 90-percent probability values ("B" values in MIL-HDBK-5) shall be used. The values used shall be consistent with overall spacecraft reliability requirements.

- The effects of thermal environment shall be included in the determination of member size. All critical combination of loads and loading shall be evaluated.

l. Factors of Safety -- The components shall comply with the following factors of safety:

Components	Limit*	Yield	Ultimate	Proof	Burst
Flight Loads: Payload Structure	1.00	1.25	1.40	--	--
Nonflight: MSGE (New Design)	1.00	3.00	5.00	2.00	--
Pressure: Vessels (Flight)	1.00	--	--	1.50	2.00

- Limit load is defined as the maximum flight load predicted by the launch vehicle contractor or generated from a current coupled loads analysis.

m. Strength Requirements -- All structural components shall be capable of withstanding the simultaneous application of the limit load, applied temperature, and all accompanying environmental phenomena without experiencing excessive elastic or plastic deformation. All structural components shall be capable of withstanding the simultaneous application of the ultimate load, applied temperature, and all accompanying environmental phenomena without experiencing failure. A factor of safety shall not be applied to any test condition except for the load. The applied load, including combined loads, shall always be less than the maximum allowed load to give a margin of safety of zero or greater.

n. Venting -- Venting of the spacecraft shall be designed so that a pressure differential greater than 1.0 psi shall not be applied across any surface of the spacecraft. Venting shall be accomplished in a manner that prevents the existence of a line of sight path between any vent and optical surfaces of sensors or of experiments and spacecraft subsystems.

o. Mass Properties -- Mass properties for each spacecraft shall be calculated and provided in the Spacecraft Alignment and Calibration Handbook.

p. Antenna Dampers -- DEB Model 1025 dampers shall have a deadband less than 9°. CCR 1787A MOD 428

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3.6.1.2 Thermal Subsystem

a. Function -- The onboard thermal control subsystem shall maintain temperature control over portions of the satellite that have mechanical or electrical characteristics adversely affected by temperature extremes. The temperatures of all thermally significant components and locations shall be monitored by flight temperature sensors; as a goal, calibrated flight temperature sensors should be accurate to within ±3°C, all other flight sensors should be accurate to within ±5°C. Onboard thermal control is required during ascent and on-orbit operation but not necessarily during test or prelaunch phase. (See 3.6.1.2.d.)

A combination of onboard and nonflight external (STE) thermal control is required during testing and prelaunch activities.

b. Configuration -- Thermal control of the spacecraft and of the instruments shall be accomplished by judicious selection and use of passive (whenever possible) or active thermal control elements. These elements may consist of any of the following: thermal blankets, selected absorption and emissivity finishes, insulators, spacecraft equipment layout, heaters, louvers with associated Thermal-Control Electronics (TCE), thermostats, and thermistors.

All thermal blankets must be outgassed before thermal vacuum exposure with the satellite or flight components or before flight if a given blanket has never been exposed to thermal vacuum testing. The blankets must be baked out at 80°C in a vacuum for 48 hours or until a quartz Crystal microbalance (QCM) held at -20°C shows an increase of less than 10 counts per hour. The IMU patch blankets are the only exceptions to the above requirements since they are not scheduled to be made until after satellite thermal vacuum testing. In this case, the blanket materials to be used for the IMU patch blankets must be outgassed as above with the other blankets. After bakeout, these materials shall be bagged and tagged and handled with white gloves when the blankets are made.

c. Survival Thermal Range -- All components installed in or on the satellite shall have a nonoperating survival temperature range of at least -5° to +45°C except that the battery shall have a nonoperating survival temperature range of 0 to +32°C and the GFE, which shall be defined in the respective Unique Instrument Interface Specifications (UIISs). Equipment shall be required to operate successfully subsequent to, but not during, exposure to survival temperature range extremes. The lower limit for the IMU shall be -l°C.

d. Operating Thermal Range -- The temperature control subsystem shall maintain the satellite operating equipment within allowable operating limits during launch and orbital operation phases. During testing and prelaunch phases primary thermal control will be onboard spacecraft, however, auxiliary equipment outside the spacecraft may be used to maintain thermal control as required. Specific instrument payload temperature limit requirements shall be as referenced in paragraph 3.1.4.2. Requirements for component qualification and acceptance thermal testing shall be as defined in Section 4.

e. Thermal Gradients -- Thermal gradients on the mounting surfaces of sensors that require accurate alignment shall be minimized so as not to cause misalignment greater than that specified in the individual instrument interface documents. Maximum rate of change of temperature at any unit mounting interface shall be less than 7°C per hour with constant electrical power input.

f. Thermal Fields of View -- Radiators and louvers shall be provided with clear fields of view to the extent required for satisfactory operation. Coolers shall be provided with clear fields of view as specified in the instrument UIISs.

g. Operating Flexibility -- The thermal control subsystem shall provide makeup heaters or other components as necessary to ensure flexibility of the subsystem to control satellite temperatures if equipment does not provide the amount of heat expected either through failure, reduced duty cycle, or being turned off.

h. Heat Flow from or to Sensors -- In general, heat exchange between instrument sensor modules and the spacecraft shall be minimized to the fullest practical extent. Allowable exchanges shall be considered instrument-dependent and shall be documented in the instrument's unique interface specification. Each instrument supplier shall be responsible

for the sensor module's thermal design and shall apply the specifics of this requirement to the conception and implementation thereof.

i. Component Selection -- As a minimum requirement, components of the thermal control subsystem shall be similar to flight-proven components. This requirement applies particularly to active thermal control elements.

j. Active Control Failure -- Failure of a control unit to actively operate a louver shall not result in equipment temperatures beyond the operational limits.

k. Propulsion System Effects

NOAA-K,L,M

The thermal control subsystem shall include protection of all equipment against thermal soakback and plume heating resulting from the hydrazine engines or from the apogee kick motor (AKM) burns. This requirement applies particularly to equipment and cables on the RSS and to equipment mounted in the ESM above the AKM. The thermal design shall consider the surface property degradation from the hydrazine and AKM contaminants noted during the launches of the previous NOAA and DMSP satellites.

NOAA-N and N-prime

Section not applicable.

l. RSS Isolation

NOAA-K,L,M

Conductive and radiative thermal isolation of the reaction control and propulsion equipment (RCPE) subsystem and other satellite components from the RSS structure shall be provided, so that heating of these components during solid rocket motor burn does not degrade the satellite performance requirements as specified in paragraph 3.2.1.1c.

NOAA-N and N-prime

Section not applicable.

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3.6.1.3 Attitude Determination and Control Subsystem

a. Boost and Ascent Mode

NOAA-K,L,M

The attitude determination and control subsystem (ADACS) shall provide the following boost mode functions:

- Satellite attitude determination from liftoff to orbit mode handoff.
- Guidance monitoring from liftoff to completion of orbit insertion.
- Attitude control of the spacecraft from separation through delta V correction.
- Determination of satellite velocity after orbit insertion so that a closed-loop delta V correction may be performed by the RCS (see paragraph 3.6.1.8).
- Provide signals to initiate isolation of hydrazine thrusters from the hydrazine supply tanks and to deplete the hydrazine in the isolated system.
- Provide attitude control signals to the RCS to effect closed-loop attitude control of the satellite during coast (after the delta V correction) to mission mode handoff.

NOAA-N and N-prime

The attitude determination and control subsystem (ADACS) shall provide the following boost mode functions:

- Attitude control of the spacecraft from separation through mission mode handoff.
- Provide attitude control signals to the RCS to effect closed-loop attitude control of the satellite, as appropriate, to mission mode handoff.
- Accept a ”ready for separation”signal from the launch vehicle.
- Initiate control signals to release the V-band.
- Initiate control signals to release deployable equipment.
- Acquire Earth.

b. Orbit Mode -- The ADACS shall provide proper orientation of the sensor axes by determination and control of the spacecraft attitude during its orbital life. The ADACS shall maintain the proper relationship between the spacecraft coordinate system (as defined in paragraph 3.6.1.li) and the reference geoid. The reference geoid shall be an ellipsoid with center at the center of mass of the Earth, having a mean equatorial radius of 6,378,145 meters and an inverse flattening of 298.25.

c. Configuration -- The ADACS functional block diagram is shown in Figure 8. The ADACS shall be a zero momentum system composed of the following or equivalent equipments configured to meet the requirements of paragraphs 3.6.1.3a, b, and d through 1.

- One Static Earth Sensor -- CO_2 band horizon sensor outputting pitch and roll attitude errors with respect to local vertical. The sensor shall be provided with redundant output data buffers.

- Digital Sun Sensor -- A single-axis solar aspect sensor that is read continually to provide yaw control loop monitoring on the ground. The data are sampled and averaged once per orbit by the spacecraft central processing unit (CPU) to provide yaw attitude and gyro bias determination.

- Four Reaction Wheel Assemblies -- Brushless, dc, ball bearing reaction wheels, including all electronics used for control of the reference axes.

- Redundant Magnetic Torquing Coils -- Air core coils interacting with Earth's magnetic field to provide momentum unloading of the reaction wheels.

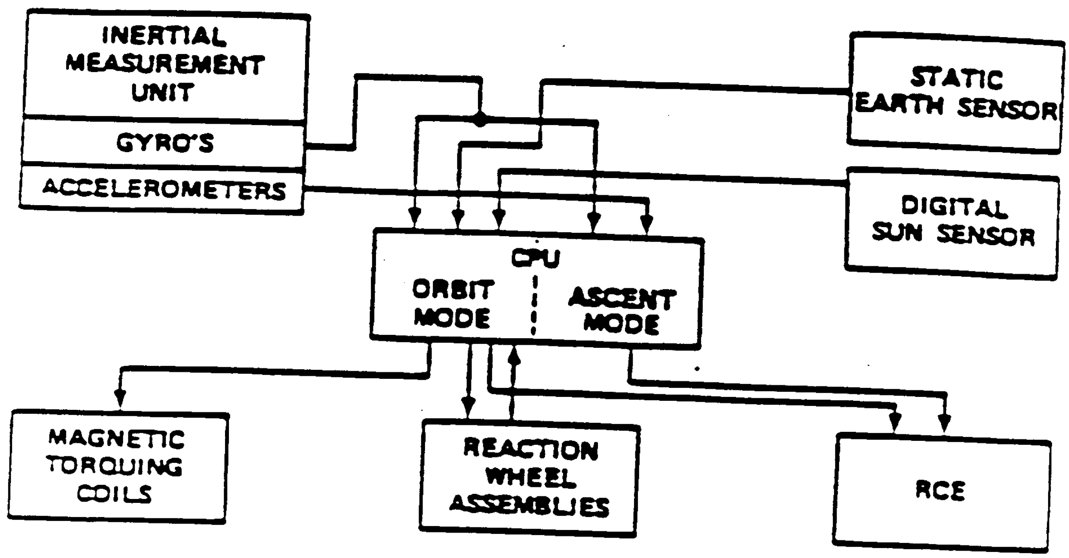

Figure 8. Attitude Determination and Control System

- One Inertial Measurement Unit -- On NOAA-K, L, and M, the IMU shall be used for determining attitude during boost, final orbital velocity during boost mode, and attitude rates in orbit. On NOAA-N and N-prime the IMU shall be used for determining spacecraft rates.

- Reaction Control Equipment (Not part of ADACS)-- As described in paragraph 3.6.1.8, the nitrogen (N_2) thrusters shall have the capability for momentum unloading of reaction wheels during orbit nominal and yaw gyro compassing (YGC) modes when the spacecraft is in Earth lock and to assist in initial acquisition if a primary wheel fails.

- Central Processing Unit (Part of command system) -- Containing the software for attitude determination and control processing to provide control to the appropriate items listed in paragraph 3.6.1.3c. Specific requirements for this software are included in paragraph 3.6.3.

d. Boost Mode Requirements -- Performance of the functions specified in paragraph 3.6.1.3a shall be accomplished by the IMU and CPU on NOAA-K, L, and M so that the orbit parameters specified in paragraph 3.2.1.lb are achieved following the delta V correction. The functions specified in paragraph 3.6.1.3a on NOAA-K,L,M,N and N-prime shall be accomplished concurrently with the required attitude maneuvers, array deployment, and antenna deployments.

e. Orbit Mode Requirements -- After completion of the ascent phase of the mission, the ADACS shall autonomously under preprogrammed direction begin closed-loop Earth acquisition procedure. After Earth lock has been achieved, and during all subsequent orbit mode operations through the satellite life, the ADACS shall maintain the attitude of the spacecraft coordinate system of paragraph 3.6.1.li to the local geodetic vertical/orbit normal coordinate system to within the following three sigma values when subjected to all external and internal disturbance torques assuming that all GFE meets UIIS requirements:

- Error (each axis) = ±0.2 deg
- Roll error rate = ±0.015 deg/sec
- Pitch and yaw error rate✧ = ±0.035 deg/sec

f. Attitude Determination -- The ADACS shall provide for determination of the attitude errors defined in the preceding paragraph (e). Determination shall be performed on board the satellite, and the determined errors shall be made available to the attitude control system for closed-loop control of the satellite. Attitude determination data shall be telemetered to the ground. The absolute orientation of the spacecraft shall be determinable on the ground to an allowable 3σ error of ±0.1 degree about the roll, pitch, and yaw spacecraft axes.

g. Acquisition -- The ADACS shall be capable of acquisition and re-acquisition beginning with any possible satellite attitude. Stable attitude control to within approximately 5 degrees per axis shall be obtained within approximately 100 minutes after commanded acquisition. Stable control to the values given in paragraph 3.6.1.3e shall be achieved within 24 hours after commanded acquisition contingent upon receipt of ephemeris data by the satellite.

h. External Disturbances -- External disturbances produced by stray magnetics, aerodynamic drag, solar pressure, and gravity gradient shall be considered in sizing the momentum unloading system.

i. Internal Disturbances and Uncompensated Angular Momentum -- Internal disturbances produced by rotating parts and impulsive motions produced by GFE payload and spacecraft bus components shall not cause position or rate errors to exceed the specified values in paragraph 3.6.1.3e.

j. Structural Interactions -- Design of the ADACS shall be such that it prevents vibration modes of any parts of the satellite, in particular the solar array, from interacting significantly with the attitude control system. The control loop parameters shall be such that sensor actuator/structural dynamics do not affect control loop stability. Interactions of vibrational modes of the solar array and other appendages shall be considered so that array deflections or attitude errors in excess of the requirements of paragraph 3.6.1.3e do not result.

k. Ephemeris Dependence -- When beginning from an initially correct ephemeris, onboard propagation for 1 week shall not yield errors that exceed:

- Spacecraft along-track position = ±50 km
- Spacecraft across-track position = ±25 km
- Sun angle = ±0.5 degree

The spacecraft does not have to meet the pointing requirements of paragraph 3.6.1.3e if the propagated ephemeris yields errors, which are caused by incorrect initial conditions or updates, less frequent than those previously mentioned.

l. Magnetic Fields

- Instrument Sensitivity -- Care shall be taken in the design to minimize any impact on instrument performance from magnetic interference.

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3.6.1.4 Command and Control Subsystem

The command and control subsystem (C&CS) shall meet the following requirements:

a. Function -- The C&CS shall control the various satellite functions in accordance with commands received from ground command and data acquisition (CDA) stations and supplemented by signals and data supplied by spacecraft subsystems. The C&CS shall receive command signals from the dual Command receiver/demodulator described in paragraph 3.6.1.5g. These signals shall be decoded in the C&CS and sent to spacecraft units as control signals.

b. Configuration -- The C&CS shall consist of a controls interface unit (CIU), a CIU Annex (CXU), two central processing units (CPUs), a signal conditioning unit (SCU), description authenticating unit (DAU) and a redundant crystal oscillator (RXO). A controls power converter (CPC), configured as shown in Figure 9 and housed within the CIU, is considered part of the power subsystem.

   The CPU shall include the main spacecraft memory and processing subunits and shall provide flexible data processing and computational capability to support all spacecraft subsystems. The CPU shall provide an elapsed time code counter and a method of presetting and correcting the elapsed time code counter.

   Specific requirements for software within the CPUs are detailed in paragraph 3.6.3.

   The CIU shall be the central element of the command and control subsystem, handling all interfaces between units within the C&CS, and all interfaces between units of the C&CS and the remainder of the satellite. The CIU shall decode messages (real-time and stored commands and CPU data), provide control signals to the spacecraft, and provide clock frequencies to the C&CS and the rest of the spacecraft as required. The CIU shall check command word length and spacecraft address and parity. A time code counter shall be supplied by the CIU to allow either CPU to determine real time. The execution of RXO primary/- backup commands shall not require the presence of the RXO clock.

   The RXO shall consist of two independent 5.12-MHz crystal oscillators, one of which shall always be selected as the oscillator source for the CIU frequency division chain. High-current control signals shall be supplied to spacecraft units by the SCU in response to low-level control signals from the CIU.

c. CPU Capability -- In addition to software required for command functions, the C&CS shall include the software for launch phase guidance and control appropriate to the launch vehicle and for attitude determination and control on orbit. The CPU shall have hardware capability to perform at least 16-bit fixed-point arithmetic. In addition, capability shall exist for performing 32-bit (double precision) arithmetic through either hardware or software implementation.

d. CPU Capacity -- Memory-size requirements shall be dictated by the exact method of functional implementation and shall be limited to 64 K words. The CPU memory shall allocate sufficient storage space in orbital operations to accommodate 2,048 stored commands plus time labels (as described in paragraph 3.6.1.4e). This space shall be readily expandable into the on-orbit space memory to provide up to 2300 commands plus labels. The expansion capability shall be demonstrated prior to spacecraft acceptance. Total memory capacity shall be sufficient for a single computer to perform all software functions covered in paragraph 3.6.3.1.

e. Time Labels -- Each stored command shall have an accompanying time label to indicate when the command is to be acted upon Each time label shall be 17 bits long. The CPU shall check for coincidence between each time label and the 17 most significant bits of the elapsed time counter. This shall result in a granularity of approximately 1 second and a period of greater than 36 hours for the time label.

f. Command Word Structure -- The uplink format shall provide for an Encrypted/Authenticated Mode, Encrypted Only Mode, and Clear Mode. Each mode shall be as depicted in Figure 10. The command word shall be in accordance with the requirements of GSFC 530-UGD-GN, Ground Network (GN) User's Guide, June 1993. The uplink

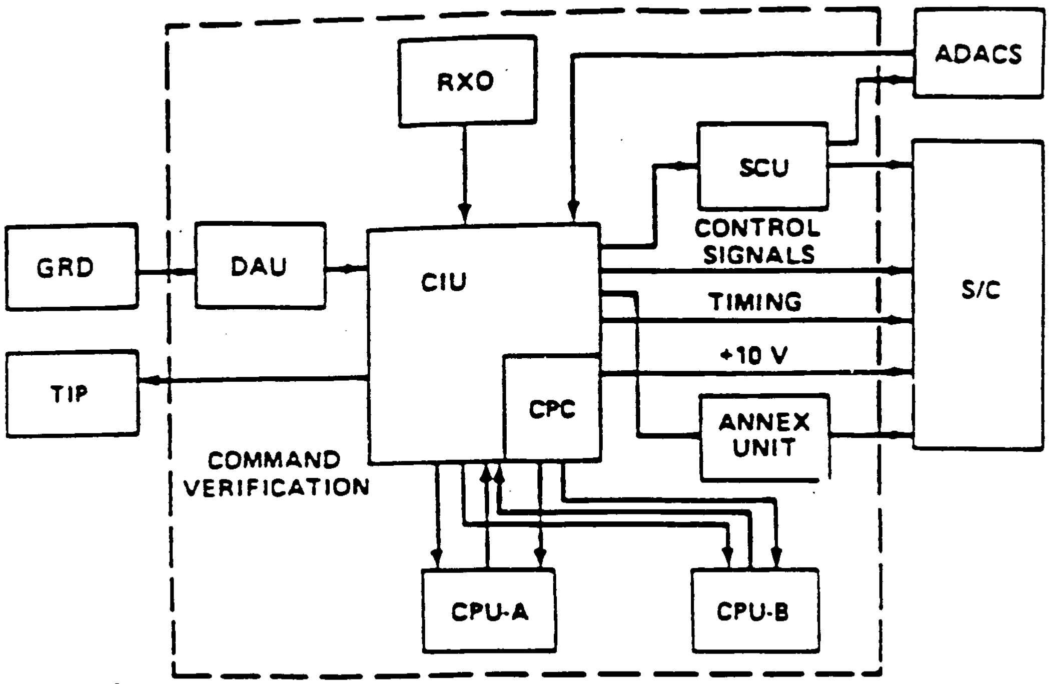

Figure 9. Command and Control Subsystem

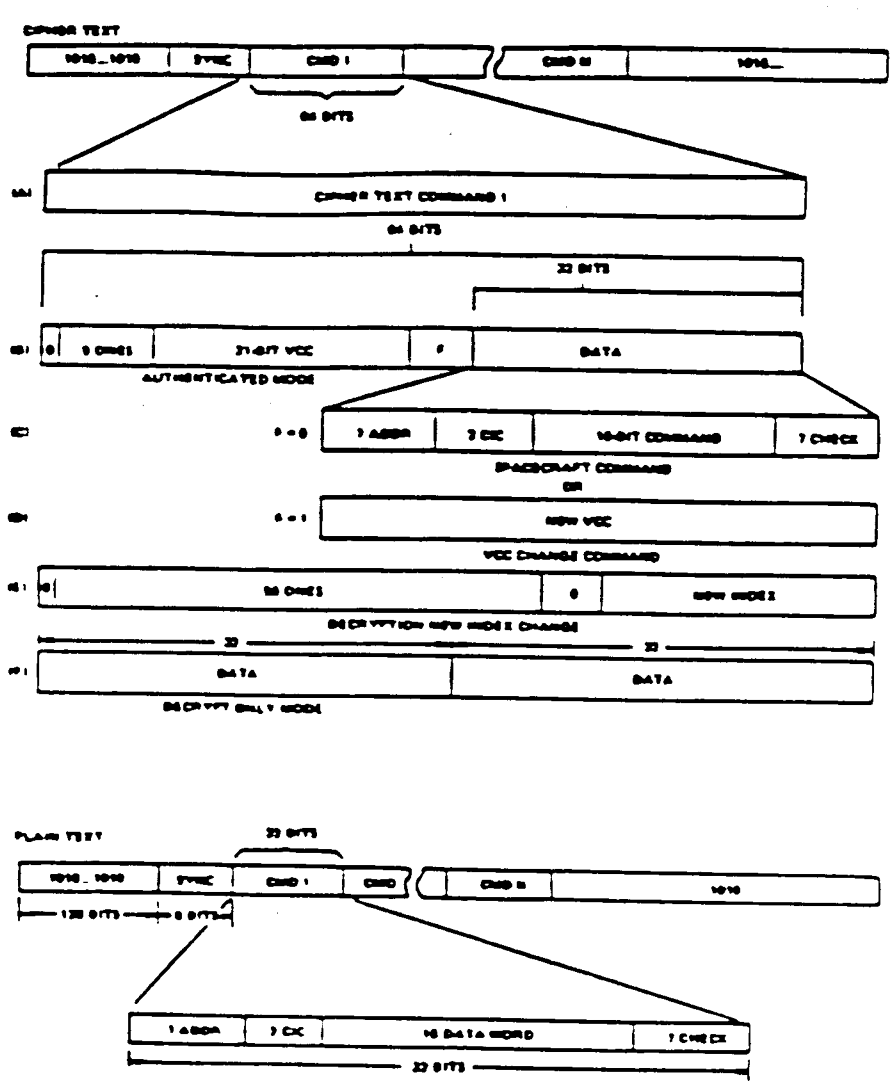

Figure 10. GSTDN Command Data Formats

security shall accommodate the KIR-23 EMS technology. The command word shall have a 7-bit spacecraft address field; a 2-bit CPU Interrupt code that shall define the data field type; a 16-bit command field which may be a CPU data word, CPU command, or CIU command; and a 7-bit check field (over the 32 spacecraft command bits). A 128-bit barker code and 8-bit sync code shall precede the command sequence.

Provision shall be incorporated to utilize a 21-bit VCC, a new VCC command, and a new index command. The format shall have a 32-bit command structure in the clear mode and a 64-bit structure in all other modes each exclusive of sync and barker code. The first word in any sequence shall define the number of commands to follow. Each command sequence shall be followed by the 1/0 pattern.

g. Uplink Security -- Uplink security shall be provided through use of an encryption and authentication system. The design of the spacecraft portion of this system shall utilize the KIR-23 EMS technology, which the contractor shall obtain directly from the National Security Agency (NSA). Means shall exist both by command and onboard timer to deactivate this system. The timer shall be designed such that any failure within the timer circuitry or other circuitry needed for timer operation (e.g., CPU, CPU software, CIU, PSE, and RXO, etc.) shall result in Clear Mode operation. No combination of failures in the decryption circuitry shall permanently affect the ability to command either spacecraft command processor. The encrypting/decrypting keys shall be protected by means of suitable security measures.

h. Memory Stability -- Contents of the spacecraft memory shall not be altered by any condition allowed on the ±10 or +28 volt regulated bus as specified inparagraphh 3.6.1.6h.

i. Control Signals -- The C&CS shall provide control signals to the various satellite units in the form of a low-voltage level (on), a high-voltage level (off), or a pulse to the low-voltage level. The signal levels and rise and fall times shall be as given in 2629668. Control signals provided to the instruments, instrument data-handling units, and critical spacecraft subsystems shall be from either CIU bus.

j. Clock -- The C&CS shall use a high stability temperature controlled oscillator package. Clock stability shall meet or exceed the following limits after stabilization:

- 5 parts in 10^9 per minute
- 1 part in 10^8 per day
- 3 parts in 10^8 per week
- 2 parts in 10^6 per year

This oscillator shall be used by the CIU to generate all timing required by the C&CS and the rest of the satellite except for timing signals supplied by the data processing subsystem described in paragraph 3.6.1.7. The oscillator shall be able to be set to its design frequency within ±0.000001 percent.

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3.6.1.5 Communications Subsystem

The communications subsystem shall meet the following requirements.

a. Function -- The communications subsystem shall receive and demodulate ground commands to the spacecraft and shall transmit spacecraft data to ground stations. Spacecraft data shall include housekeeping telemetry and instrument data both recorded and in real time. The data shall be as defined in paragraph 3.6.1.7. One of the carriers on which the spacecraft data are transmitted shall be usable by a CDA station for autotracking. The communications subsystem shall include an antenna for receiving UHF data that shall be distributed to the DCS and SARP receivers by use of a diplexer. The communications subsystem shall also include a two-element nested antenna capable of receiving SARR data and an L-band SARR transmitter antenna. There shall be no data-inversion uncertainty resulting from any communications link. NOAA-K, L, M, N, and N-prime shall transmit data whose phase polarities are identical with those of NOAA-J. SARR phase polarity shall be controlled by the GFE supplier.

b. Configuration

- Equipment -- The contractor provided portions of the communications subsystem shall include three data transmitters in the meteorological S-band (STX1, STX2, and STX3), one launch telemetry transmitter in the Air Force Satellite Control Facility (SCF) band (STX4), two VHF realtime APT transmitters (VTXs), two VHF beacon transmitters (BTXs), a dual command receiver/demodulator, three S-band antennas (SBAs), a VHF real-time APT antenna (VRA), four S-band omni-telemetry antennas (SOAs), two S-band omni-command antennas, a beacon antenna (BA), a UHF data collection system antenna (UDA), a diplexer (DPD) for coupling received signals to both the DCS and SARP, a SARR receiver antenna (SRA), a SARR L-band transmitter antenna (SLA), and any required RF filters (RFFs) and switch assemblies (RFSs) configured as shown in Figure 11.

- Communications Links -- The equipment of the preceding paragraph shall form 11 communications links between the spacecraft and the ground. These links shall be:

  - Command Link -- Ground commands transmitted by a CDA station shall be received on the S-band command antenna and demodulated by the command receiver/demodulator. Demodulated command signals shall be sent from the communications subsystem to the C&CS. Both sides of the dual GRD shall operate continuously. The GRD design shall contain no single point failure modes that would result in a catastrophic failure of the GRD.

  - VHF Beacon Link -- The 8.32-kbps split-phase TIP output data stream, including analog and discrete housekeeping telemetry and low-rate payload sensor data, shall be transmitted continuously in real time by one of the redundant VHF beacon transmitters using the beacon antenna.

  - APT Link -- Reduced resolution geometrically corrected AVHRR data on a 2.4-kHz subcarrier and ancillary data as formatted by the MIRP shall be transmitted continuously in real time by one of the redundant VHF APT transmitters to worldwide APT stations.

  - HRPT Link -- Full resolution AVHRR, AMSU, and TIP data formatted in the MIRP at 0.66-Mbps split phase shall be transmitted continuously in real time by one of two S-band transmitters to worldwide HRPT stations. The HRPT link shall be capable of using either the lowest (1698 MHz) or highest (1707 MHz) transmitters of the three meteorological S-band transmitters. Capability shall be provided for HRPT data at the 1702.5 MHz frequency in a degraded mode using an LCP S-band antenna or an RCP SOA.

  - CDA-Stored Data Link -- AVHRR, AMSU, and TIP data as formatted in the MIRP and stored on one or more tape recorders shall be transmitted at 2.66-Mbps nonreturn to zero (NRZ) or 1.33-Mbps split phase using up to two S-band transmitters to a CDA station during a suitable CDA pass. Either of two MIRP

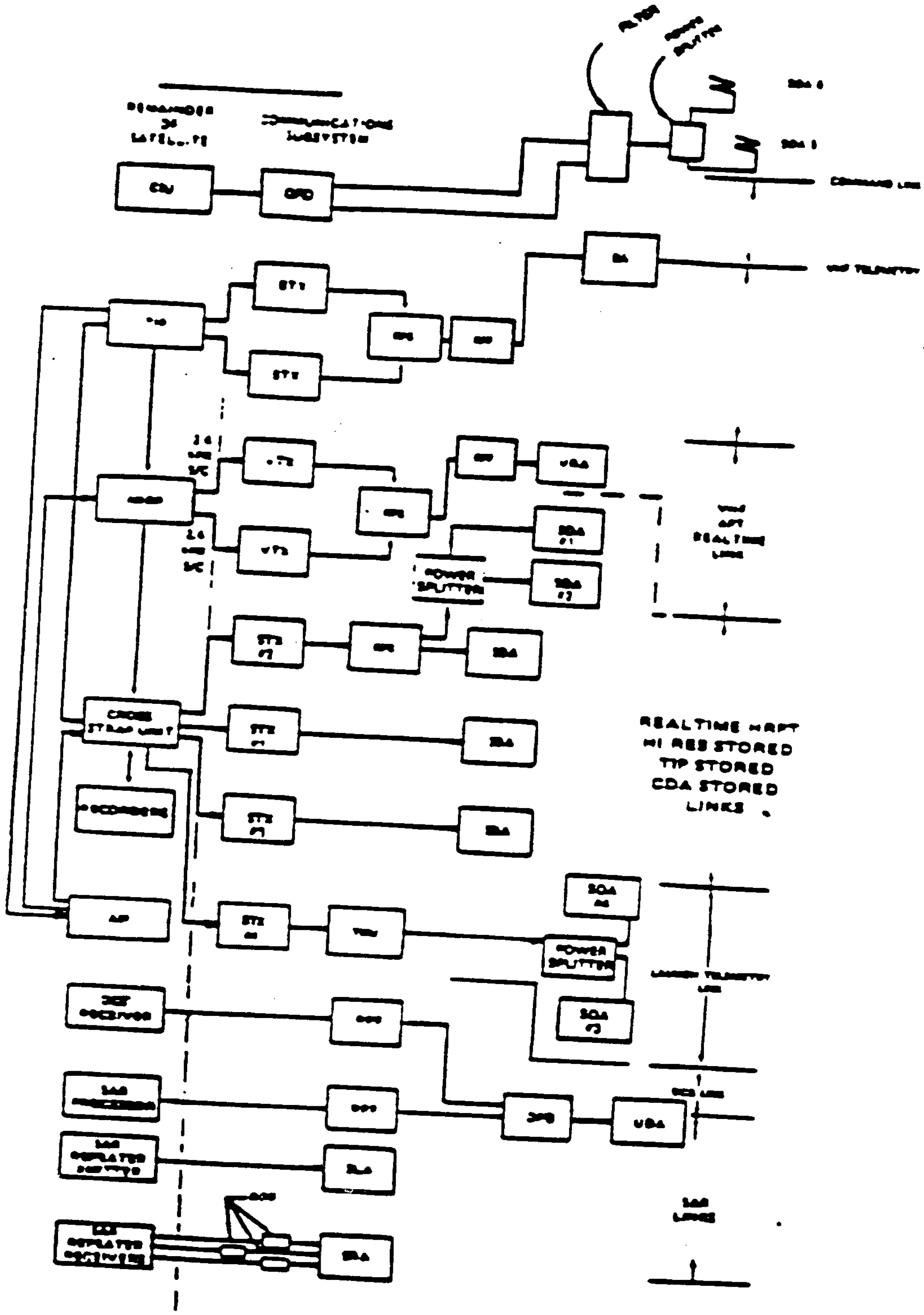


Figure 11. Communications Subsystem

data streams can be so processed; namely, continuous global coverage (reduced resolution) AVHRR, AMSU data with submultiplexed TIP data or up to 10 minutes per orbit of local area coverage (identical with "HRPT link" preceding, but NRZ, and recorded). The CDA-stored data link shall be capable of using any of the four meteorological S-band transmitters.

- TIP/AMSU-Stored Data Link -- The stored TIP/AMSU output data stream at 0.33 Mbps split phase including analog and discrete housekeeping telemetry and low-rate sensor data shall be stored and played back on one of the four meteorological S-band transmitters during orbits that have no CDA contact but that have suitable contact with the European ground station. The TIP/AMSU-stored data link shall be capable of using any of the four meteorological S-band transmitters. The tape recorder will be played back by stored command

- S-band Beacon Links -- When the satellite is in the specified orbit, two S-band links shall be available as beacons transmitting real-time TIP or AIP data. One of the links shall use the 1702.5 MHz S-band transmitter. The other link shall use the S-band launch telemetry transmitter. The option shall exist of operating either or both of the above links, including operating both continuously.

- DCS Link -- Data from balloon platforms and remote automatic observation stations throughout the world shall be received by the communications subsystem's UHF DCS antenna and sent to the DCS receiver (which is part of the instrument payload).

- S-band Launch Telemetry Link -- TIP boost mode telemetry data stream at 16.64-kbps split phase shall be transmitted continuously by the S-band transmitter (STX-4) through an S-band omni-antenna (SOA #3 and SOA #4) during launch to mission mode handoff.

- SAR Receive Link -- Three frequencies transmitted from ground-based platforms associated with the SAR instruments for retransmission to SAR ground stations.

- L-band SAR Link -- SAR processed and unprocessed repeater data modulated onto a single L-band carrier by the SAR repeater transmitter.

c. Operational Modes

- Normal Operations -- The communications subsystem shall be capable of simultaneous operation of the VHF beacon link, the DCS link, the SAR links, the VHF real-time link, and the HRPT real-time link with up to two highrate S-band playback links or with the TIP or AIP S-band playback link.

- Launch Operation -- The USAF Satellite Control Facility (SCF) compatible launch and early orbit emergency communication system shall be capable of operating the S-band launch telemetry link and transmitting TIP boost mode telemetry data at 16.64 kbps split phase during launch mode.

- Beacon Only Operation -- Either S-band beacon link, the one in the meteorological S-band or the one in the SCF compatible S-band, can be operated in any spacecraft emergency in boost or orbit mode in addition to the VHF beacon link.

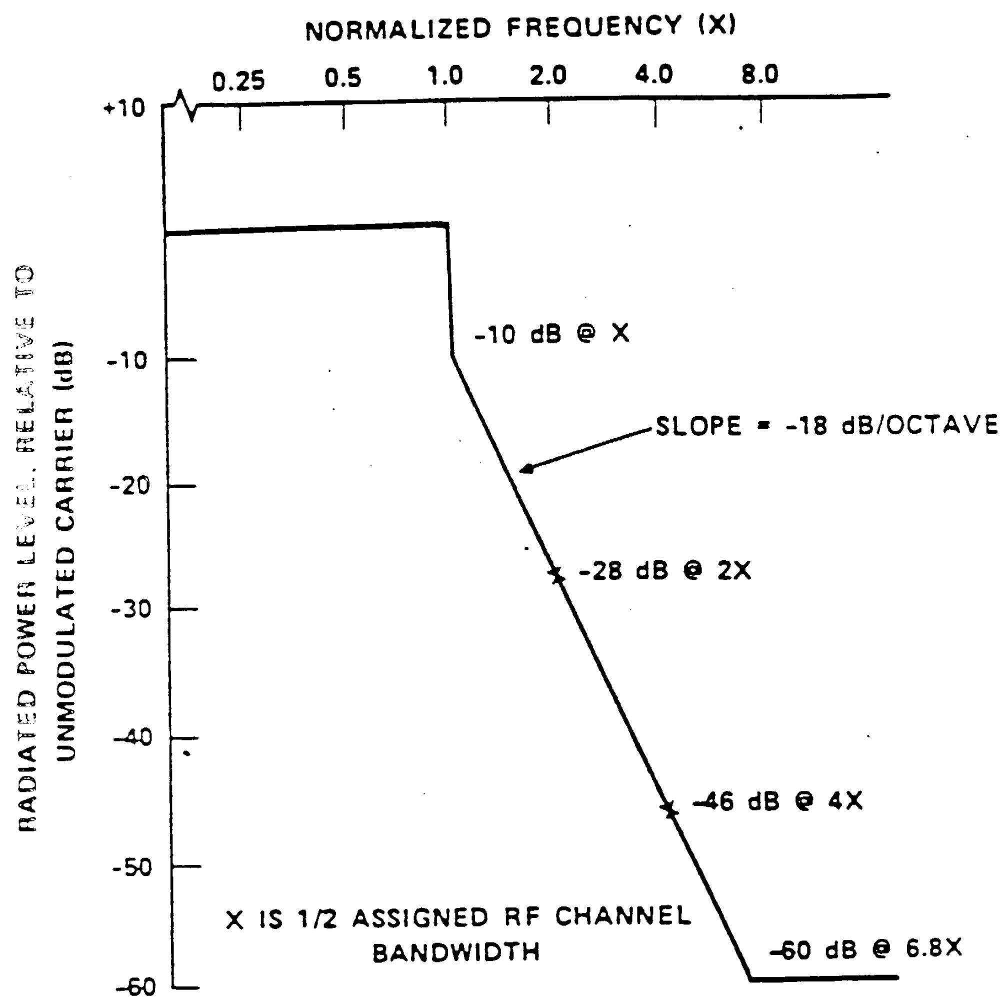
d. Meteorological S-band Link -- The meteorological S-band link shall have the characteristics given in Table 5 and Figure 12.

- Format -- The input signal will be any one of the following:

  - NRZ Pulse Code Modulation (PCM) bit stream at a rate of 2.6616 Mbps. Input data rate stability will be ±0.2 kbps.
  - Split-phase bit stream at a rate of 0.3327 Mbps.

Table 5 Meteorological S-band Link Characteristics

Frequency Band	1695 to 1710 MHz
Carrier Frequencies	1698, 1702.5, and 1707 MHz
Frequency Stability	±0.002%
Out-of-Band Emissions Transmitted Bandwidth	See Figure 12 and paragraphs 3.6.1.5j and n
Modulation Index	PM 2.35 radians ±5% for lowest bit rate
Modulation Rate, Type	2.6616-Mbps, NRZ, 0.6654-Mbps, split phase 0.3327-Mbps, split phase 1.33-Mbps, split phase
EIRP	+36.3 dBm for 1698, +37.5 dBm for 1702.5, and +36.0 dBm for 1707 MHz



SERVICE	X (kHz)
S BAND (PER CHANNEL)	3500 kHz
APT	25 kHz
VHF BEACON	30 kHz

NOTE: The requirements of Paragraphs 3.6.1.5j and n must also be met.

Figure 12. Radiated RF Signal Level

- Split-phase bit stream at a rate of 0.6654 Mbps.
- Split-phase bit stream at a rate of 1.3308 Mbps.

The modulation index shall be 2.35 radians ±5 percent for the lowest bit rate.

- Spurious Phase Modulation -- The phase jitter under condition of no modulation nor vibration shall not exceed 3 degrees rms when measured in a phase lock-loop bandwidth of 30 Hz.

- Phase Linearity -- The phase modulation transfer function shall be such that the RF phase versus input voltage curve, over the range of 2.5 radians (peak-to-peak), shall be linear within ±0.25 radian of the straight line through the end points.

- Amplitude Modulation -- A digital signal producing a phase deviation of 2.35 radians peak-to-peak shall cause less than 5-percent amplitude modulation of the STX output carrier.

- Modulation Transfer Performance -- When modulated with an 8.32 KHz symmetrical (50% duty cycle) square wave, the demodulated waveform from a transparent phase modulation (PM) detector shall meet the requirements specified herein. MOD 381 CCR1650

  - Rise and Fall Times -- The rise and fall times shall each be less than 200 nanoseconds.

  - Overshoot -- The overshoot shall be less than 25 percent of the steady-state peak-to-peak step.

  - Bit Asymmetry -- When measured through a point midway between the peak steady-state output voltages, the bit asymmetry of a 2.6616-Mbps data stream (using an alternating one-zero pattern) shall be less than 2 percent prior to the transmitter modulator and 5 percent overall. Bit asymmetry is defined as the maximum bit length minus the average bit length, divided by the average bit length. MOD 381 CCR 1650

- Turn-on Time -- Within 6 seconds after application of dc power, each transmitter shall meet all specification requirements through the specified temperature range and over the specified supply voltage range.

e. VHF Beacon Link -- The beacon link shall have the characteristics given in Table 6.

- Incidental Amplitude Modulation -- Incidental amplitude modulation (AM) shall not exceed 5 percent when the digital signal produces a phase deviation of 2.35 radians peak-to-peak.

- Phase Linearity -- The phase modulation transfer function shall be such that the RF phase versus input voltage curve, over the range of the 2.5 radians (peak-to-peak), shall be linear within 0.25 radian of the best straight line through the end points.

- Modulation Transfer Performance -- When modulated with an 8.32KHz symmetrical (50% duty cycle) square wave, the demodulated waveform from a transparent PM detector shall meet the requirements specified herein.

  - Rise and Fall Times -- The rise and fall times shall each be less than 30 microseconds.

  - Overshoot -- The overshoot shall be less than 25 percent of the steady-state peak-to-peak step.

  - Bit Asymmetry -- When measured through a point midway between the peak steady-state output voltages, the bit asymmetry of the 8.32-kbps data stream shall be less than 3 percent. Bit asymmetry is defined as the positive pulse length minus the negative pulse length, divided by the sum of the positive and negative pulse lengths.

- Spurious Phase Modulation -- Phase jitter under the conditions of no modulation and no vibration shall not exceed 3-degree rms when measured in a phase lock-loop bandwidth of 30 Hz.

Table 6 Beacon Link Characteristics

Frequency Band	137 to 138 MhZ
Carrier Frequency	137.35 and 137.77 MHz
Frequency Stability	±0.002%
Out-of-Band Emissions Transmitted Bandwidth	See Figure 12 and paragraphs 3.6.1.5j and n
Modulation Rate and Type	PM ±67? max at 8.32-kbps split phase
EIRP (worst-case)	Orbit mode

f. APT Link -- The VHF APT link shall have the characteristics given in Table 7.

- Modulator Input -- The VTX shall have one modulator input consisting of an amplitude modulated 2.4-kHz subcarrier. Modulation index of the subcarrier shall not exceed 92 percent.

- Deviation Linearity -- The frequency deviation of the VTX must be within a band ±600 Hz about the best fit straight line over the deviation range of -18.7 to +18.7 kHz. In addition, the frequency deviation of the transmitter must be within a band ±100 Hz about the best fit straight line for each 1 kHz segment of the 0 to ±18.7 kHz deviation range.

- Frequency Response -- The amplitude versus frequency characteristic of the modulator shall be flat within ±0.5 dB from 0.1 to 4.8 kHz.

- Frequency Stability -- The transmitter shall operate at the specified center frequency with a stability of ±0.002 percent.

- Residual Frequency Modulation -- Residual FM of the carrier shall not exceed 340 Hz peak-to-peak deviation in the absence of vibration.

- Amplitude Modulation -- The transmitter output shall not exhibit amplitude modulation greater than 5 percent when the deviation is ± kHz.

g. Command Link

- Center Frequency -- The exact pre-assigned center frequency (F_c) of each S-band receiver shall be 2026.000 MHz. Receiver frequency instability shall not exceed 0.003 percent of center frequency.

- Command Signal Modulation Parameters -- The GSTDN system shall utilize binary phase-shift-keyed (NRZ-M) signal phase modulating a 2026.0 MHz carrier. Table 8 indicates the modulation parameters to be used with the GSTDN command link. Figure 13 shows the modulation waveform of the digital commands. The waveform in Figure 13 phase-modulates a 2026 MHz RF carrier. The GSTDN Command Receiver/Demodulator (GRD) shall receive the phase-modulated carrier, demodulate the carrier to achieve the Binary Phase Shift Keyed (BPSK) signal, and then demodulate the BPSK signal into the command bits. These shall then be sent to the decryption authentication unit.

- Command RF Sensitivity and Dynamic Range -- The command data outputs of the GRD shall be capable of delivering data with an error rate of 1×10^{-6} or less whenever the received EIRP is between -35 and -75 dBm over 90 percent of 4 π steridians and also whenever the spacecraft is Earth oriented and the received EIRP is between -35 and -87 dBm.

- Squelch -- The S-band receivers shall incorporate a squelch circuit that inhibits the command demodulator circuits whenever the command signal level at the input ports of the receivers falls below the squelch level defined as follows. Define the nominal threshold level as the lowest level at the input ports of the receivers that meets the command RF sensitivity and dynamic range requirements previously stated. The squelch level shall be between the nominal threshold level and 7 dB below the nominal threshold level.

- Clock Bandwidth -- Activation of command outputs shall occur within 20 milliseconds after application of a modulated RF carrier. Deactivation of command outputs shall occur within 20 milliseconds after removal of modulation of the RF carrier.

Table 7 APT Line Characteristics

Frequency Band	136 to 139 MHz band
Carrier Frequencies (NOAA-KLM)	137.50 to 137.62 MHz
Carrier Frequencies (NOAA-N/N-prime)	137.10 and 137.9125 MHz
Frequency Stability	±0.002%
Out-of-Band Emissions Transmitted Bandwidth	See Figure 12 and paragraphs 3.6.1.5j and n
Modulation Rate and Type	±17 kHz FM with a 2.4 -kHz subcarrier
EIRP	+33.5 dBm

Table 8 GSTDN Command Link Modulation Parameters

Parameter	Specification
Carrier Frequency	2026.0 MHz
Carrier Modulation Index	1.0 ±0.1 rad, peak
Subcarrier Frequency	16.0 kHz
Bit Rate	2.0 Kbps

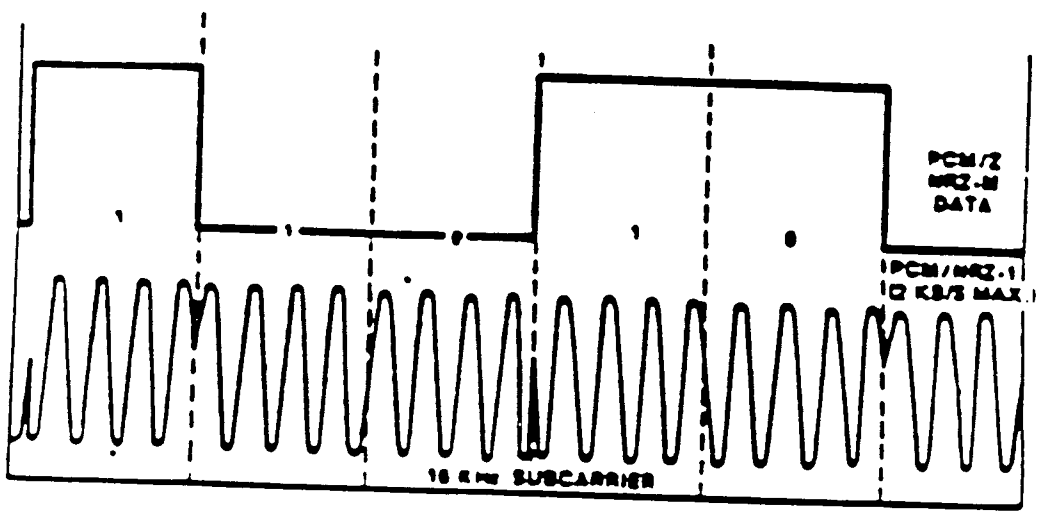


Figure 13. Typical Digital Command and Associated Modulation Waveform

h. Data Collection System Link -- The communications subsystem shall provide a net gain as referenced to the data collection receiver RF input connector to a linearly polarized plane wave having an arbitrarily oriented polarization vector that is within the upper and lower bounds defined in Figure 14.

Net gain is defined to include antenna gain relative to a linear isotropic reference, line insertion loss, RF filter insertion loss, voltage standing wave ratio (VSWR) loss, and diplexer loss. The calculated VSWR loss shall be based on the DCS receiver that has a maximum input VSWR of 1.5 to 1.

The design shall not depend on in-band attenuation which increases system noise temperature in order to meet the gain requirements herein.

i. Antenna Interference -- Antennas shall be mounted so as not to obstruct the field of view of any sensor or active/passive cooler during all aspects of the mission, including antenna deployment. Nor shall any antenna interfere with the deployment or rotation of the solar array.

j. Antenna Conducted Outputs -- Spacecraft transmitted emissions (including SAR) shall not degrade the spacecraft command receiver performance beyond specification. Antenna conducted outputs at the DCS-2 and A-DCS-3 receiver antenna input shall not exceed the values shown in the following table.

Maximum signal level (dBm)	Frequency MHz
0	1-15
-20	15-375*
-60	375-385
-100	385-396
-125	396-401.570
-145	401.570-401.730
-125	401.730-406
-100	406-411
-60	411-425
-20	425-1000
0	1000-10000

*The signal level at DCS-2 and A-DCS-3 receiver input terminal shall not exceed -60 dBm at the following frequencies:

31.0808 MHz $\pm$ 40 kHz
339.4885 MHz $\pm$ 40 kHz

k. Antenna Polarization -- All communications subsystem antennas shall be designed to transmit (or to receive, as appropriate) right-hand circularly polarized signals, except for the antenna dedicated to the S-band transmitter operating on the center channel of the allocated S-band range (1702.5 MHz), the S-band launch telemetry (1702.5 MHz), the beacon antenna, and the SAR L-band antenna. To reduce adjacent channel interference while allowing HRPT data to be transmitted on the 1698 MHz and 1707 MHz transmitters simultaneously, the polarization of the antenna dedicated to the (1702.5 MHz) S-band transmitter shall be left-hand circularly polarized. The beacon antenna may be linearly polarized. The 1702.5 MHz launch telemetry antenna polarization is characterized in the following paragraph. The SARR L-band antenna shall be left-hand circularly polarized. The ellipticity of the antennas associated with the GFE shall be as defined in the appropriate UIISs. The SOAs shall not be required to meet a specific ellipticity specification. The VRA shall have an ellipticity of no greater than 10 dB with a goal of 6 dB. The SBAs shall have an ellipticity of no greater than 6.0 dB with a goal of 4.5 dB. The above ellipticity requirements are for the respective antennas when mounted on the full-scale antenna test model.

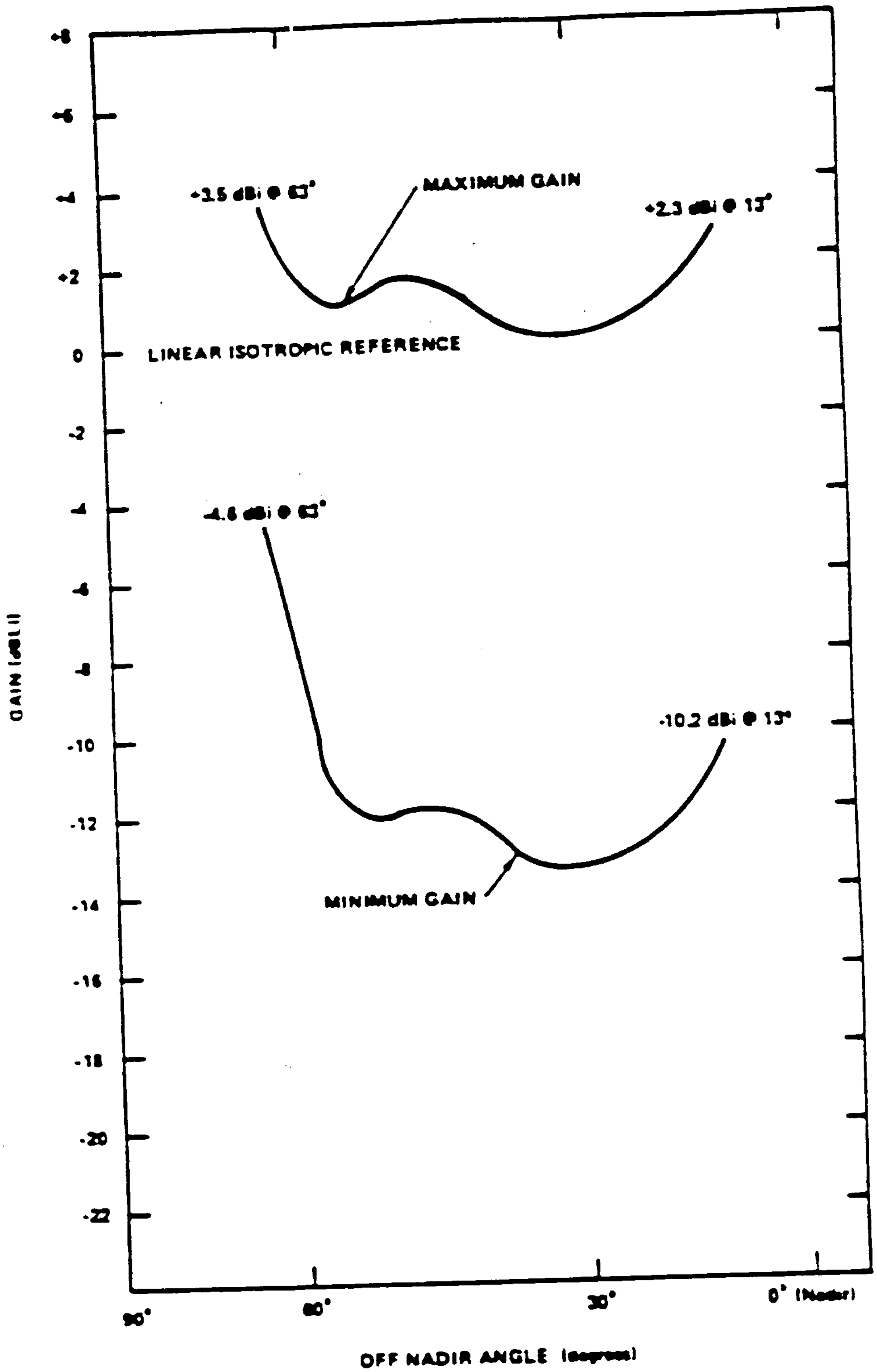


Figure 14. Net Gain Required at DCS Receiver RF Input Terminal

l. Coverage Pattern -- The directional S-band transmitting antennas, the VHF APT transmitting antenna, and the UHF receiving antenna serving the DCS link shall provide 63° half-angle conical coverage in orbit. The antennas serving the Search and Rescue links shall provide 60° half-angle conical coverage in orbit. The VHF beacon antenna shall radiate a beacon EIRP greater than +9.5 dBm over 90 percent of a 4 π steradians in the mission mode. The beacon antenna shall not require deployment mechanism devices. All antenna specifications shall be met when the antenna is mounted on the spacecraft as configured for normal operation of the antenna. The 1702.5 MHz S-band omni-antenna shall radiate a predominately right-hand circularly polarized pattern about the +X axis (nadir) and a predominately left-hand circularly polarized pattern about the -X axis (antinadir). This antenna shall appear to radiate a signal strength greater than +24 dBm over at least 90 percent of 4 π steradians when observed by an optimum polarization diversity ground receiver. The launch telemetry antenna shall provide launch mode spacecraft telemetry to launch-site receivers and Airborne Range Instrumentation Aircraft (ARIA) after fairing separation. The command antenna shall receive commands from either CDA station in any spacecraft orientation including unstable attitude control.

Prior to launch, when the spacecraft is mounted to the launch vehicle, all RF S-band, VHF Beacon, and command links shall have a means provided to test them, including receipt of data through the GFE links to GSE receivers and transmission of data to the command receiver. In addition, transmission of commands to the spacecraft command receivers and receipt of the S-band launch telemetry signal will be possible during the entire countdown and launch.

Testing and use of RF links while the spacecraft is on the booster will be contingent upon a GFE fairing design that contains re-radiators or RF transparent sections.

m. Transmitter Protection -- All transmitters shall be capable of operating into a short-circuit or open-circuit termination of any phase without damage.

n. Out-of-Band Transmissions -- In addition to the out of band transmission requirement previously specified, the following special provisions shall apply:

- Power Flux Density in the 1665 MHz and 1720.5 MHz Radio Astronomy Band -- To afford adequate protection to the radio astronomy observations being made in the bands 1660 to 1670 MHz and 1720 to 1721 MHz, the power flux density as measured at the Earth's surface for all conditions and modes of modulation shall not exceed, as a design goal, -236 dBW/m^{2}Hz. The design goal relates to the power flux density that would be obtained for all angles of arrival under assumed freespace propagation conditions. The design goal applies equally to the continuous power flux density that would be obtained for all angles of arrival under assumed freespace propagation conditions. The design goal applies equally to the continuous and discrete components of the power flux density.

- Out-of-Band Energy in the 1694.3 to 1694.7 MHz GOES Band -- The contractor shall configure the modulation of the downlinks in a manner that minimized the out-of-band energy in the Geostationary Operational Environmental Satellite (GOES) data collection reply channel (1694.3 to 1694.7 MHz). The goal is not to degrade the pertinent GOES channel by more than 1 dB except when the satellites defined by this specification are directly in the main beam of the GOES receiving antenna.

o. S-band Launch Telemetry Link and S-band Beacon Link (SCF Compatible STX-4 System) -- The S-band launch telemetry link of the SCF compatible launch and early orbit emergency communication system shall have the characteristics given in Table 9.

The requirements of paragraph 3.6.1.5d shall apply to the S-band launch telemetry link except that the bit asymmetry requirements shall apply to 16.64 Kbps split-phase data.

Normally, the STX-4 system will not be utilized when the search and rescue system is operating; therefore, this system does not have to meet the SAR emission requirements.

Table 9 S-band Launch Telemetry and S-band Beacon Link Characteristics

Carrier Frequency	2247.5 MHz
Frequency Stability	±0.002%
Modulation Type	2.6616 Mbps, NRZ 0.6654 Mbps, split phase 0.3327 Mbps, split phase 1.33 Mbps, split phase 8.32 Kbps, split phase 16.64 Kbps, split phase
EIRP	+24 dBm (over 95% of sphere)
Polarization	RCP/RCP

The STX-4 system need not operate simultaneously with the 1702.5 MHz S-band omni-telemetry system during on-ground testing.

The STX-4 transmitter must be provided with a ground cooling system which will be used for ground testing and prelaunch activities.

The capability to use either 1702.5 MHz or the 2247.5 MHz telemetry link during pre-launch activities must be maintained.

All four S-band transmitters will be normally capable of simultaneous operation in orbit.

- S-band Beacon Link (in the Meteorological S-band) -- The S-band beacon link shall have the characteristics given in Table 10. Operation of the S-band beacon link shall be mutually exclusive with operation of the 1702.5 MHz S-band link in Table 5. The requirements of paragraph 3.6.1.5d shall apply to the S-band beacon link except that the bit asymmetry requirements shall apply to 8.32-Kbps and 16.64-Kbps split-phase data.

p. Search and Rescue Receive Links -- The communications subsystem shall provide four separate RF inputs to the search and rescue equipments. A 121.5-MHz receiver input shall be provided as defined in Figure 15. A 243-MHz receiver input shall be provided as defined in Figure 16. A 406.05-MHz receiver input shall be provided as defined in Figure 17. A 406.025-MHz receiver input derived from a diplexer connected to the ultrahigh frequency data collection system antenna (UDA) shall be provided as defined in Figure 18. The VSWR provided by the spacecraft to each of the four receiver inputs shall be less than 1.6:1 across a ±500 kHz band centered at each of the receive frequencies. The isolation between any two ports shall be greater than 20 dB. For purposes of VSWR loss cal culations, a maximum VSWR of 1.5:1 shall be assumed for the SAR receivers. Lossy devices, which increase the system noise temperature, shall not be used unless specifically approved by the Tiros Project Office.

q. Search and Rescue L-band Link -- The communications subsystem shall provide an antenna gain including all losses as shown in Figure 19 for a 10 watt transmitter at a frequency of 1544.5 ±0.5 MHz. The VSWR shall be less than 1.5:1 across the above band. The SAR transmitter VSWR will be no greater than 1.5:1.

r. Antenna Conducted Outputs -- Spacecraft transmitter emission shall not exceed those given in Table 11 at the SAR receiver terminals.

The contractor shall define the maximum limits for the out-of-band emission spectrum components for the SAR transmitter so that these levels can be met.

Table 10 S-band Beacon Link Characteristics (in the meteorological S-band)

Carrier Frequency	1702.5 MHz
Frequency Stability	±0.002%
Modulation Type	2.35 rad ±5%, split-phase-level PCM 8.32 kbps (orbit; 16.64 Kbps (boost)
EIRP	+24 dBm (over 90% of sphere)
Polarization	RCP/LCP

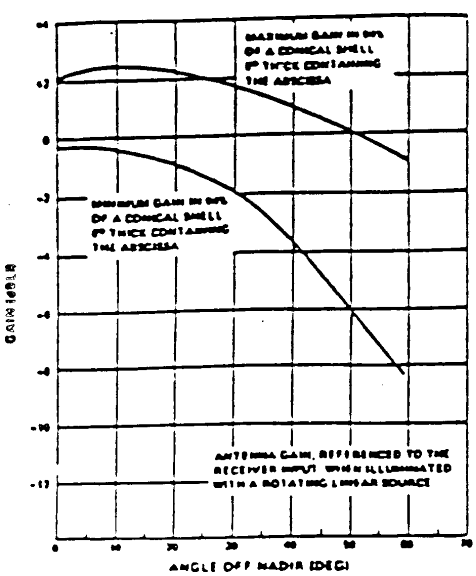


Figure 15. SARR 121.5-MHz Receiver

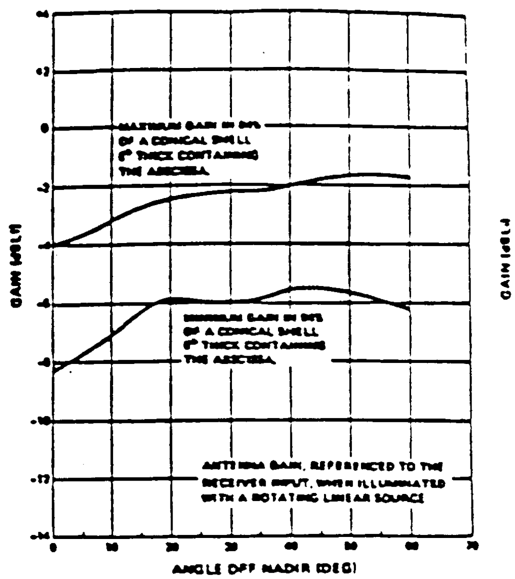


Figure 16. SARR 243-MHz Receiver

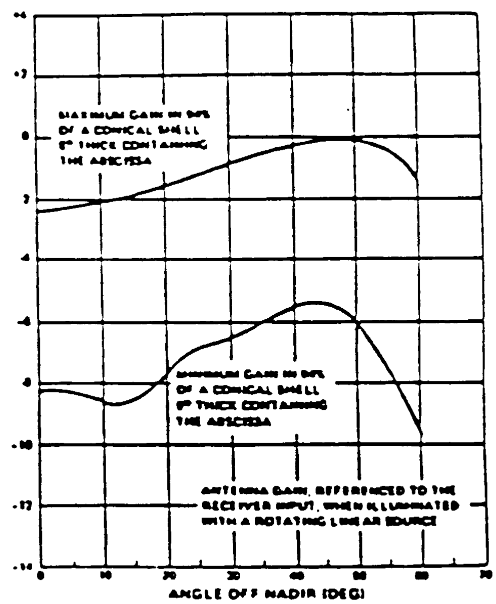


Figure 17. SARR 406.05-MHz Receiver

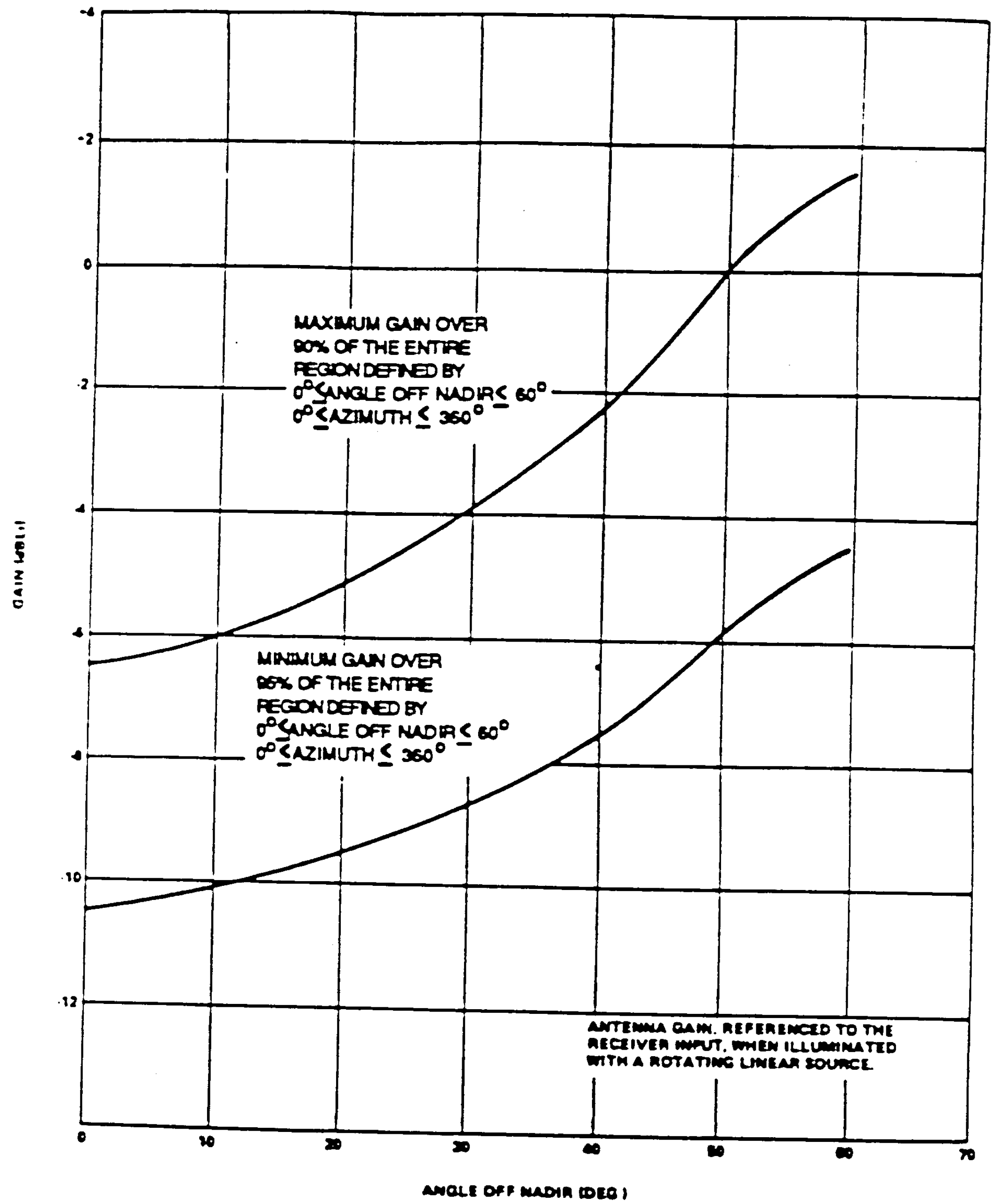


Figure 18. 406.025-MHz (SARP) Receiver Input

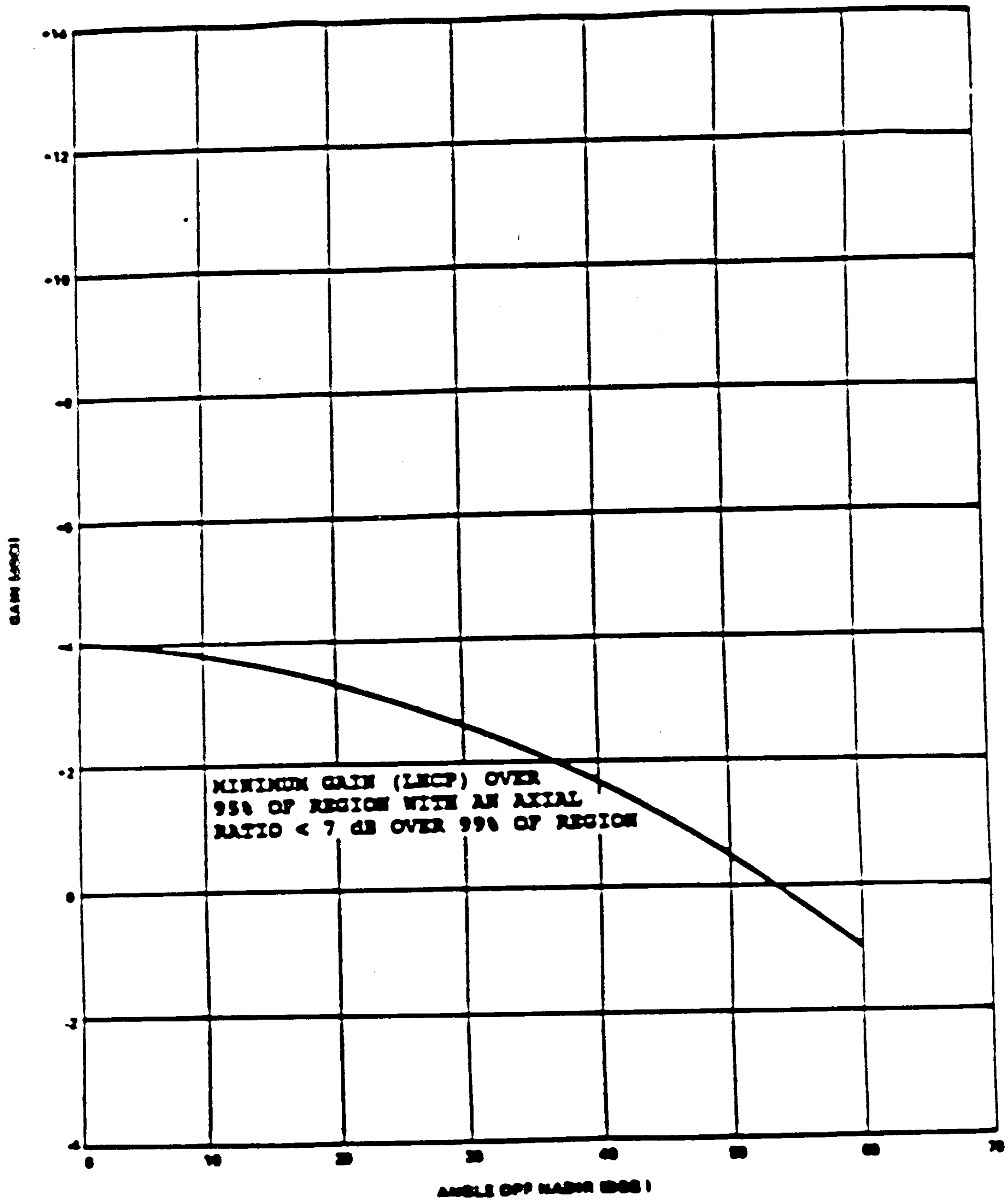


Figure 19. Antenna Gain for 10-watt 1544.5-MHz Transmitter as seen at the SARR Transmitter Output Port

Table 11 Spurious Signal Limits at SAR Receiver Inputs

Center Frequency	Frequency Band (MHz)	Limit Maximum (dBm)
SRR 243 MHz	1 to 15 15 to 228 228 to 236 236 to 240 240 to 242.925 242.925 to 243.075 243.075 to 246 246 to 250 250 to 258 258 to 1000 1000 to 10000	-20 -60 -100 -125 -145 -125 -100 -60 -20 0
SARR 121.5 MHz	1 to 15 15 to 114 114 to 118 118 to 120 120 to 121.45 121.45 to 121.55 121.55 to 123 123 to 125 125 to 129 129 to 1000 1000 to 10000	0 -20 -60 -100 -125 -145 -125 -100 -60 -20 0
SARR 406.050 MHz	1 to 15 15 to 375 375 to 385 385 to 401 401 to 405.9 405.9 to 406.2 406.2 to 411 411 to 425 425 to 435 435 to 1000 1000 to 10000	0 -20 -60 -100 -125 -145 -125 -100 -60 -20 0

Antenna conducted outputs at the SARP-2 and SARP-3 receiver antenna input shall not exceed the values shown in the following table:

Maximum signal level (dBm)	Frequency MHz
0	1-15
-20	15-375*
-60	375-385
-100	385-401
-125	401-405.970
-145	405.970-406.130
-125	406.130-411
-100	411-425
-60	425-435
-20	435-1000
0	1000-10000

*The signal level at the SARP/receiver input terminals shall not exceed -60 dBm at the following frequencies:
-31.4192 MHz ± 40 kHz
-343.2115 MHz ± 40 kHz

3.6.1.6 Electrical Power and Distribution Subsystem

a. Function -- The satellite shall provide conditioned electric power during the launch, separation, operational, and occult phases of the mission as specified in the following paragraphs.

b. Launch Phase -- The satellite shall provide internal battery power for telemetry and command and selected components of the satellite system from switch over to satellite internal power until the solar array starts providing power. Prior to liftoff, the satellite shall use ground power provided through the launch vehicle umbilical from a contractor supplied ground power supply located at the launch complex. Starting at fairing jettison, the stowed solar arrays shall provide power to these loads and recharge the batteries to the extent provided, as dictated by Sun angle incidence during the ascent phase of flight. The batteries shall provide power during the shadow period and when array power is not sufficient because of low Sun angle incidence.

   A one time maximum battery depth of discharge, normally occurring during the launch phase of the mission, shall be less than 60 percent.

c. Deployment Phase -- The satellite shall provide power through the signal conditioning unit for the operation of pyrotechnics during the ascent phase of flight.

d. Operational Phase -- The satellite shall provide conditioned power for operation of all satellite systems and instruments simultaneously. The minimum power available for GFE loads (instruments, tape recorders, and GFE contingency) shall be equal to or greater than 450 watts orbital average over a gamma angle of 0 to 80 degrees for normal operating conditions for up to 2 years in orbit. The satellite shall be capable of providing continuous peak load power of 980 watts for a minimum of 10 minutes (using batteries to supplement the arrays).

e. Occult Phase -- For the occult phase, depth of discharge shall be limited to a maximum of 28.3 percent with the full complement of batteries on line.

f. Satellite Bus Voltages -- The satellite shall provide the following bus voltages:

- 28 ±.56 V -- Switched telemetry bus
- 28 ±0.56 V -- Primary power main bus; for use by all satellite subsystems and instruments

- 28 ±0.56 V -- Primary power pulse load bus; for use by stepper motors and other satellite subsystems and the GFE instruments with large transient power requirements

- 10 ±0.5 V -- Power for telemetry and command interface circuits

g. Configuration -- The electrical power subsystem derives power from the solar array that tracks the Sun by rotation around the satellite Z-axis. The solar-array tracking angle (the angle between the Sun line and the array normal vector projection on the X-Y plane) shall not exceed ±10 degrees. Energy storage is provided by three adequately sized electrochemical storage batteries that are discharged in parallel. The batteries provide electrical power during occult portions of the ascent phase and peak loading that exceeds the solar array capability.

The output power from the solar array and batteries is regulated to provide a main bus voltage of 28 volts, a pulse load bus voltage of 28 volts, and a 10-volt bus. When excess power is available from the solar array, the battery chargers are enabled to recharge the batteries. Each battery has separate charge control circuits and is normally charged at a current limited rate with temperature compensated voltage control.

The power system shall consist of the following components as shown in Figure 20.

- Solar array panels
- Power supply electronics
- Electrochemical storage batteries
- Shunt regulators
- Battery charge assemblies
- Solar-array drive electronics
- Solar-array drive
- Controls power converter
- Battery Reconditioning Units

- Solar Array -- The solar array shall contain solar cells that have the following characteristics:

  - N/P silicon
  - 2.02 by 4.04 cm
  - 2 ohm-cm nominal resistivity
  - 13 percent minimum average efficiency
  - 10 mil thickness

- Power Supply Electronics -- The power supply electronics (PSE) shall perform the following major functions for the power subsystems:

  - Provide a commandable battery discharge disconnect function for each battery with interlock so that all batteries cannot be disconnected at the same time.
  - Provide an input for external power during the prelaunch phase for solar-array simulation.
  - Provide a 16-level V/T clamp in the battery charging circuitry.
  - Provide a 4-level current limit in the battery charging circuitry.
  - Provide for individual current limiting for each battery during charging except that only two batteries can be in high charge rate at a time.
  - Provide the following functions with redundancy: battery boost regulators, battery charge control circuits, and mode controllers.
  - Provide a 28-volt main bus with ±2 percent regulation.
  - Provide telemetry signals of battery voltage to TIP for each battery for undervoltage sensing by the satellite computer.
  - Provide diagnostic telemetry for boost channel.
  - Provide commandable battery charger circuitry for switching each battery into/out from the charge circuits(s).

- Provide failure detection circuits to automatically enable the redundant functions.
- Provide a 28-volt pulse load bus.

- Batteries -- Three adequately sized electrochemical storage batteries shall be provided to supply electrical power during occult portions of the ascent phase and peak loading exceeding the solar array capability. The batteries shall meet the requirements specified in GSFC approved Battery Cell Performance Specification, Battery Performance Specification and Statement Of Work. Battery temperature telemetry signals shall be provided to TIP for each battery pack.

  Cells that are tested by the cell manufacturer must be matched by normalizing the test data. This is accomplished by designating one cell group as baseline and by correcting individual cell data of the remaining groups by the ratio of the baseline average and the corresponding group average. It is desirous that all three flight batteries of one spacecraft meet the following, but, as a minimum, all cells of a single battery

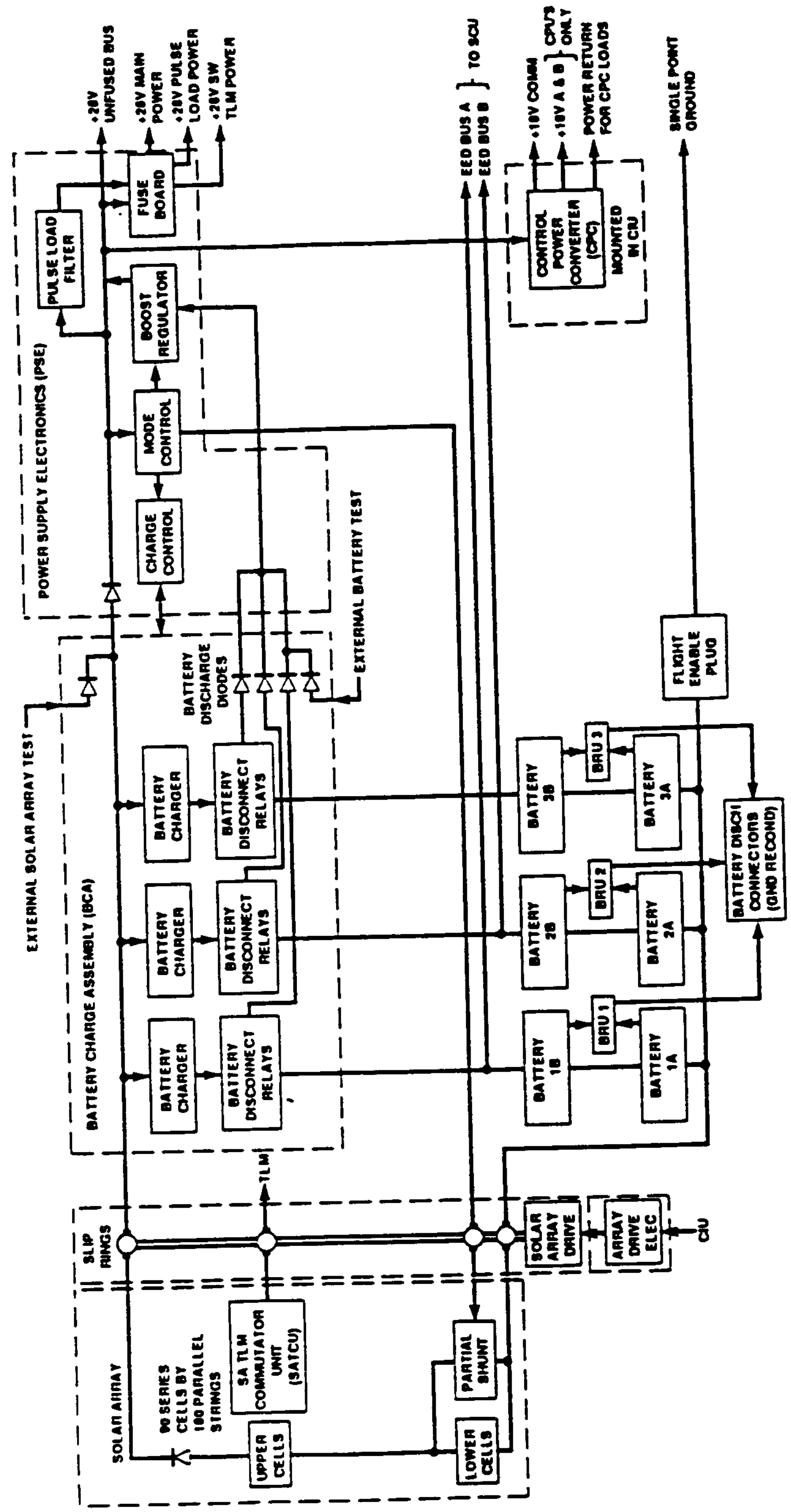


Figure 20. Power System Functional Block Diagram

shall contain plates, separator, and electrolyte from the same manufacturer's lot. Cells of a single battery shall be: a) matched to within ±3 percent of the room temperature (21°C to 25°C) and 0°C capacities and b) all cell voltages shall be within ±0.008 volt at the end of the room temperature (21°C to 25°C) and 0°C charge when tested by the cell manufacturer.

If the battery thermal control system utilizes both louvers and heaters, the thermal control nominal louver and heater set points shall be +2.0^{0}C and 0^{0}C respectively. If only heaters are utilized then the nominal heater set point shall be +4.0^{0}C.

The orbital average cell temperatures within each battery shall be maintained within a temperature range which supports the minimum battery cycle requirement (3.6.1.6h) for normal three battery operations. The minimum battery cycle requirement will also be satisfied when operating with one battery off-line due either to a battery failure or in order to support battery reconditioning operations.

When operating in the normal three battery mode, the maximum gradient between any two cells within the battery, consisting of two packs, shall be less than 6^{0}C over two years of spacecraft mission life.

During ground test operations with flight batteries installed on the spacecraft, the maximum temperature of the batteries when operated electrically shall be, as a goal, limited to +28^{0}C, but not to exceed +30^{0}C.

A battery handling plan is to be submitted to GSFC.

- <u>Shunt Regulators</u> -- The shunt regulators shall be designed for a dissipation of 16 watts maximum per unit. The shunt regulators shall regulate the main bus by drawing excess current from the lower section of the solar array. Each solar array shunt drive transistor shall have a network to supply backup collector power.

  Each shunt drive transistor junction temperature shall be within maximum derated values.

- <u>Battery Charge Assembly</u> -- Each battery charge assembly shall provide current for charging the satellite batteries. Four charge rates, including 0.5A, selectable by command shall be provided. The contractor shall choose the three remaining charge rates and submit the values to NASA for approval. The charger characteristics shall be constant current with charge current taper to maintain constant voltage as a function of temperature.

- <u>Solar Array/SAD/ADE</u> -- The solar array shall, in orbit, form a plane canted with respect to the satellite at such an angle as to provide sufficient power to the satellite through slip rings over the Sun angle range of 0 to 80 degrees. The orientation of the rotating array shall be such as to cause no interference with the fields of view of any sensor, cooler, or antenna. Proper orientation of the solar array (with respect to the projection of the Sun and array normal vectors in the orbital plane) shall be maintained to ±10° about the pitch axis by the solar array drive/array drive electronics that shall rotate the array in such directions and speeds as to allow capture and tracking of the Sun.

- <u>Controls Power Converter</u> -- The controls power converter shall accept primary regulated power at +28.O volts dc and provide +10 volt regulated power to the satellite units for interface power and to the central processing units as required.

h. <u>Performance</u>

- <u>+28 Volt Primary Bus</u> -- The power subsystem shall supply the satellite with power regulated to +28.O ±0.56 volts. The voltage on this bus shall not exceed +38.O volts nor drop below +16.O volts including all ripple, transients, and failure mode switching. The +28 volt bus output impedance shall be less than O.2 ohm at all frequencies below 100 kHz, excluding the fuses. Voltage ripple (including a maximum load induced current ripple of 0.5 A) on the +28 volt bus will not exceed 150 millivolts peak-to-peak for frequencies up to 100 kHz (200 millivolts to include deadbands). The maximum ripple current fed back from the GFE payload will not exceed 275 mA of the O.5 A maximum.

Transients on the +28 volt bus shall not exceed the following limits:

Peak Transient Amplitude (Volts)	Transient Width (micro-sec)
0.1	200-500
0.5	150-200
0.75	0-150

- +28 Volt Pulse Load Bus -- The power subsystem shall supply a pulse load bus regulated to +28 ±0.56 volts for use by stepper motors, instrument heaters, and any other loads that would cause the +28 volt primary bus to exceed ripple or transient specifications. Voltage on the pulse load bus shall not exceed +38 volts nor drop below +15 volts. The small signal source impedance of the pulse load bus shall not exceed 0.2 ohm at all frequencies below 100 kHz, excluding the fuses. Voltage ripple on the pulse load bus shall not exceed 200 millivolts peak-to-peak for frequencies up to 100 kHz, exclusive of the following repetitive transients.

  Repetitive transients on the pulse load bus caused by a load of 2.1 A, 35-msec pulse shall be less than 600 mV peak-to-peak at rates up to 14 Hz. Repetitive transients on the pulse load bus caused by 6.0 A step loads shall be less than 1.6 volts peak-to-peak. High frequency transients shall not exceed the limits given in the previous paragraph titled "+28 Volt Primary Bus" in paragraph 3.6.1.6h.

- 10 Volt Buses -- The controls power converter located within the CIU (in the command and control subsystem) shall provide +10 volts for use within the command and control subsystem and shall provide a +10 ±0.5 volts common bus to provide up to 0.8 A to power interface circuits for the satellite and instruments. The voltage on the +10 volt common bus shall not exceed 15 volts nor drop below 9 volts. The output impedance at the source shall not exceed 1 ohm at all frequencies below 10 MHz; voltage ripple shall not exceed 250 mV peak-to-peak for frequencies up to 10 MHz; voltage transients shall not exceed 1 volt, zero to peak with a maximum pulse width of 50 microseconds.

- Fusing -- All satellite fuses shall be contained within the PSE. All fuses to be derated according to PPL-18. A fuse load chart shall be provided showing rate load derating.

- Battery Charge Control -- When array current exceeds load demand, battery charging shall be controlled to
- occur in one of three modes. During the current-limiting mode, battery charge current shall be limited as specified in ±Battery Charge Assembly,± in paragraph 3.6.1.6g. Automatic switchover to a voltage limiting mode shall occur as a result of battery voltage/temperature sensing. The battery charge control shall provide sixteen, GSFC-approved, commandable temperature compensated voltage clamp levels selected from battery
- cell characterization tests or appropriate manufacturers data with GSFC approval. Charge current limits shall be selectable by ground command.

- Battery Cycling -- The battery shall be capable of meeting power subsystem battery requirements for a minimum mission life of 12,000 orbital charge/discharge cycles.

- Solar Array Construction -- The solar cell array shall consist of multiple strings of silicon N-on-P solar cells interconnected by a low resistance interconnect material (tab or mesh) and bonded to a panel structure. A coverglass with ultraviolet rejection and antireflection characteristics shall be provided as cell protection.

- Distributed Power Returns -- Power distributed from the power subsystem shall be returned via the harness to a single point on the satellite. All power returns (+28, +28 pulse load, and +10) shall be isolated from each other and from satellite structure within the boxes and the satellite. The 10-volt and 28-volt pulse load power

  returns shall go back to the source (CPC or PSE) and from there to the single point ground. Other loads shall return directly to the single point ground. The contractor shall connect them to minimize cross-talk among loads .

- Accessibility -- The satellite shall provide access for placement and removal of power disconnect plugs during satellite integration, test, and launch phases. Provisions shall be made for charging the satellite batteries during test and launch phases without removing the batteries from the satellite.

- Primary Power Overload Protection -- The satellite power system shall provide protection, except for the CIU, SCU, and IMU, from overloads or short circuits that should occur in any subsystem or instrument. The RWAs and both sides of the ADE shall each be individually fused. A main bus or pulse load bus short circuit failure of any protected instrument or subsystem, as previously defined, shall not result in loss of that power system bus once the short has been removed.

- Main Bus and Pulse Load Bus Under/Over voltage Protection -- The satellite shall provide a means to transfer to backup power system functions for a voltage below 27 volts for a time greater than 1.5 seconds or for a voltage greater than 30.5 volts for 50 msec maximum. The bus should not drop below 16 volts during under-voltage nor shall it exceed 38 volts for over voltage.

- Battery Reconditioning -- The Power System shall have the capability of reconditioning batteries during orbital operation. Provisions shall exist for the reconditioning and temperature monitoring of flight batteries while mounted on the satellite prior to and after mating to the launch vehicle.

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3.6.1.7 Data-Handling Subsystem

a. Functions -- The data-handling subsystem shall collect and process data from all onboard instruments, as well as col-- lecting all housekeeping telemetry, for transmission to the ground. Data processing shall include formatting and resolution adjustment. The subsystem shall record data as required and provide data outputs in real time, as well as playback outputs at high rates.

b. Configuration -- The data-handling subsystem shall consist of the following components:

- Tiros Information Processor (TIP)
- Manipulated Information Rate Processor (MIRP)
- AMSU Information Processor (AIP)
- Cross Strap Unit (XSU)
- Digital Tape Recorders (DTRs) for KLM
- Solid State Recorders (SSRs) for MNN'

The functional arrangement and flow among these components are shown in Figure 21. Figure 21a presents the KLM data-handling system. Figure 21b illustrates the N and N' data-handling system.

c. Performance -- MIRP shall process and format AVHRR data and multiplex AMSU and TIP data. The AIP shall interface with the three AMSU instruments for the NOAA-KLM satellites and with the AMSU-A1, AMSU-A2, and MHS instruments for the NOAA-NN-prime satellites and provide data to the MIRP and XSU. TIP shall format all other downlink data. The DTRs or SSRs shall record and play back selected MIRP and TIP/AMSU data. The XSU shall provide switched interfaces among TIP, AIP, MIRP, DTRs (or SSRs), and transmitters. MIRP shall output automatic picture transmission (APT) data for real-time transmission by one of the VHF transmitters. MIRP shall output high resolution picture transmission (HRPT) data for real-time transmission by an S-band transmitter. MIRP shall output local area coverage (LAC) data for selective recording and global area coverage (GAC) data for continuous recording. TIP shall output TIP data to MIRP for formatting into the LAC, HRPT, and GAC data. The AIP shall provide AMSU data to the MIRP and combined TIP and AMSU data to the XSU. TIP shall output the TIP data for continuous transmission by one of the VHF beacons and by the SCF-compatible S-band beacon. TIP data shall be available on command for realtime transmission by one of the transmitters in the meteorological S-band and for recording. The TIP shall provide a data stream to the AIP. Playback data from DTRs or SSRs shall be output to transmitters in the meteorological S-band on command, with up to two being played back simultaneously.

d. Interfaces -- Interfaces shall conform to the following documents:

- TIP, MIRP, AIP, and XSU interface with each other and with the spacecraft shall conform to 2629668.

- All interfaces with the payload sensors shall conform to IS-3267415 and the instrument/GFE unique specifications listed in Table 1.

e. TIP -- TIP shall format all downlink data except for data from the AVHRR, AMSU, MHS, and SAR. It shall synchronize the payload sensors other than the AVHRR, AMSU, and SAR, control the data outputs, and accept CPU data, command verification data, and housekeeping telemetry data. It shall add synchronization, identification, and time code overhead and shall output the data to the BTX, MIRP, AIP, and the XSU in accordance with mode commands received from the CIU. TIP shall be redundant. TIP shall perform in accordance with the detailed requirements of the following subparagraphs:

- Modes -- One half of the redundant TIP shall operate whenever spacecraft power is applied; either half can be selected by command, and TIP performs in the following modes:

  - Boost Mode -- In boost mode, TIP shall provide the overhead data, command verification, CPU-B telemetry data, all housekeeping data, and launch mode telemetry at high rate.

  - Mission Mode -- In mission mode, TIP shall provide the overhead data, command verification, CPU data, housekeeping telemetry, and all sensor data except for AVHRR and AMSU data.

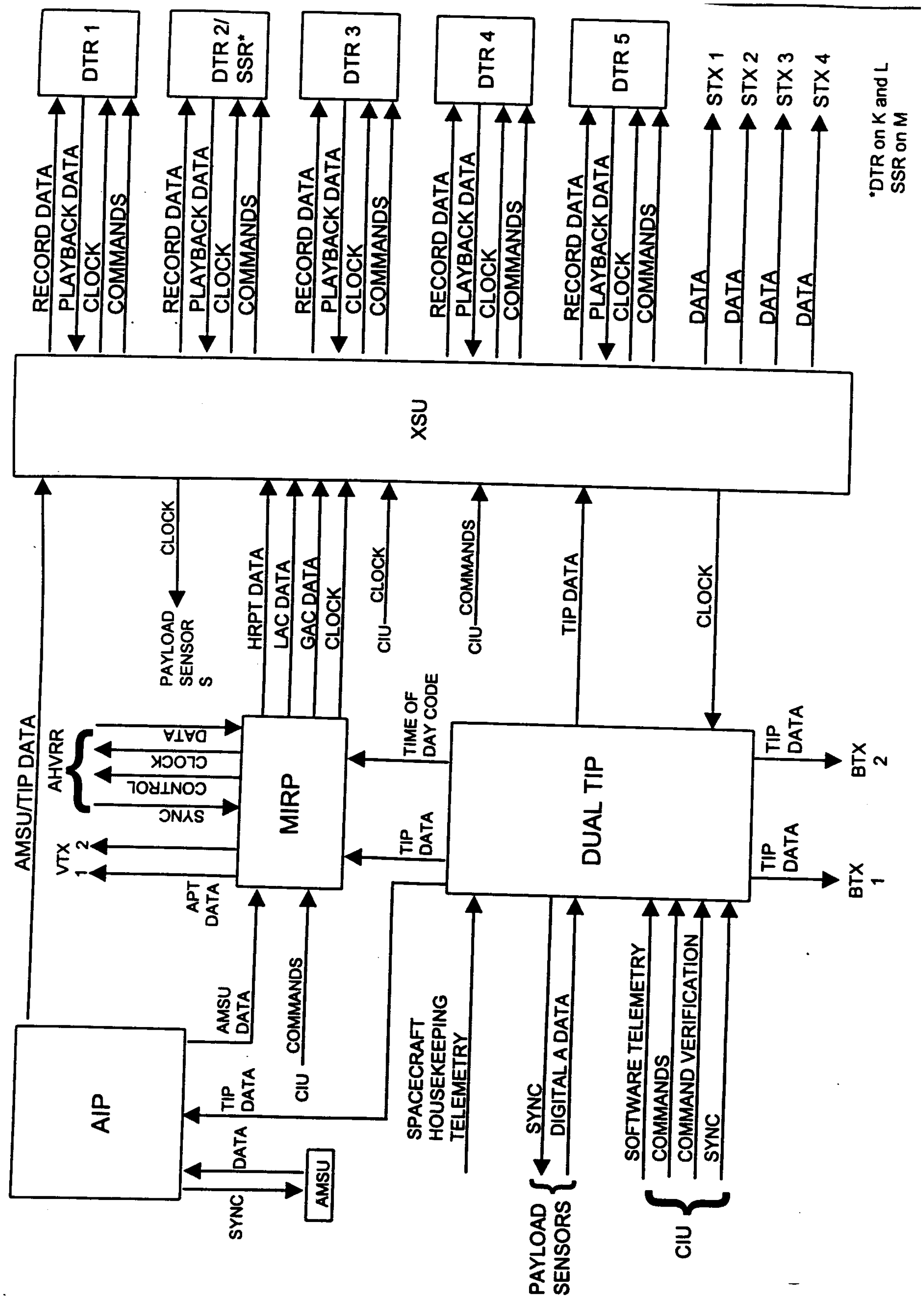

Figure 21a. KLM Data-Handling System

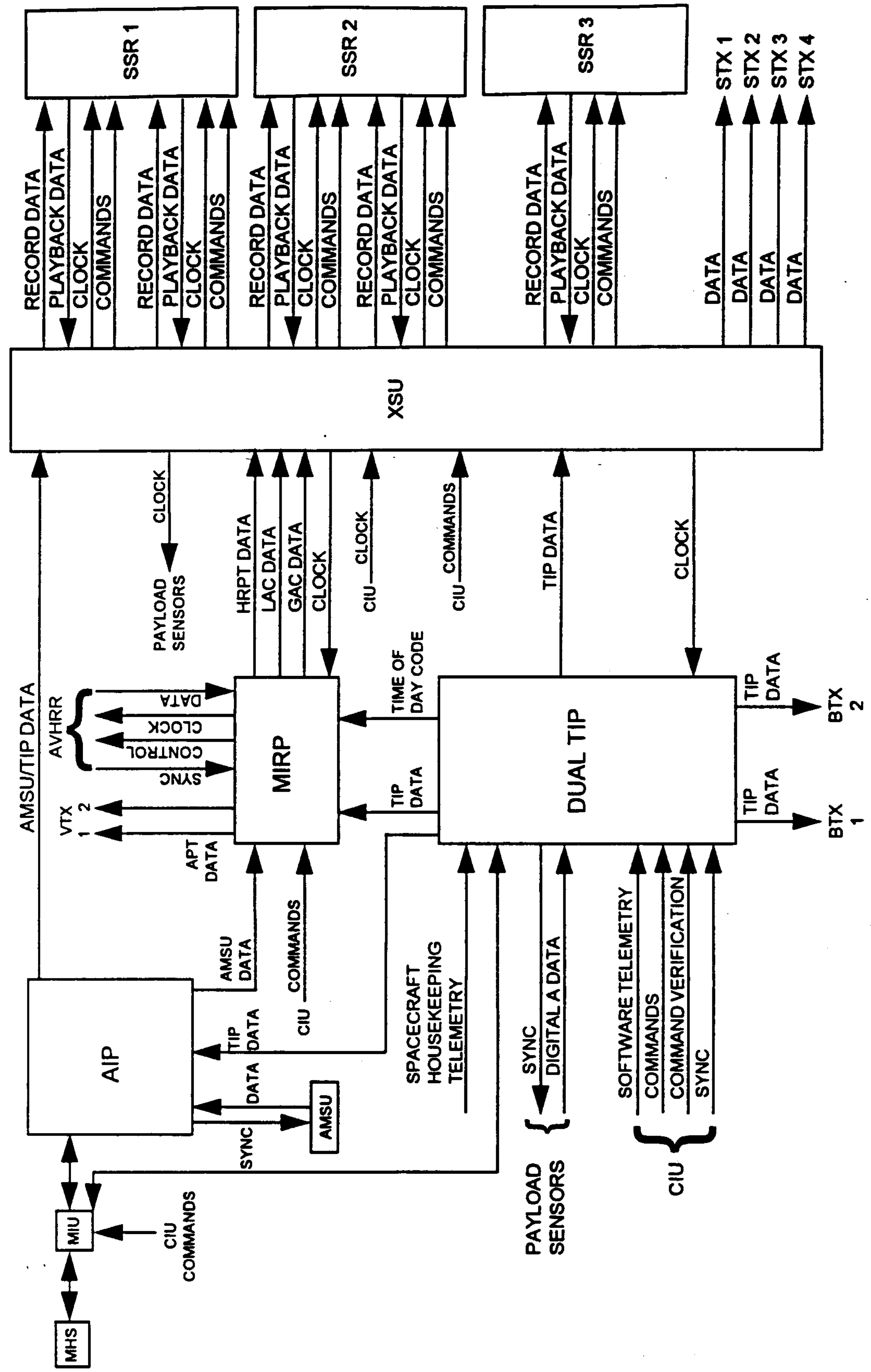

Figure 21b. NN' Data Handling System

- Dwell Mode -- In dwell mode, TIP shall provide the overhead data, command verification, and a single channel of housekeeping telemetry. The channel shall be selected by command.

- CPU Dump Mode -- In CPU dump mode, the TIP shall provide a modified mission mode format. The modification shall dedicate all CPU-A and CPU-B word locations to either CPU-A or CPU-B dump data in accordance with a command.

• Output Format -- The TIP output shall be serial digital words with the most significant bit first. The rates and lengths shall conform to the following requirements.

	Boost Mode	Mission, Dwell, & CPU Dump Mode
Bit Rate	16.64 kBps	8320 bps
Word Length	8 bits	8 bits
Minor Frame Length	104 words (0.05 sec)	104 words (0.1 sec)
Major Frame Length	320 minor frames (16 sec)	320 minor frames (32 sec)

The format shall meet the requirements in the following subparagraphs:

- Boost Mode Format -- Each minor frame of the boost mode format shall contain a synchronization code, spacecraft ID, command verification, time code, CPU-B telemetry, six words of housekeeping telemetry, and DAU status. The format shall be fixed with time code data appearing during frame "0."

- Mission Mode Format -- Each minor frame in mission mode format shall contain synchronization code, mode status, major frame and minor frame identification, dwell address identification, command verification, six words of housekeeping telemetry, spacecraft ID, CPU telemetry flag, command verification flag, parity check, CPU-A and CPU-B telemetry, DAU status, and payload sensor, (digital-A) data (except for AVHRR, MHS, and AMSU data). The format containing CPU telemetry and payload sensor data shall be controlled by a read only memory (ROM) within the TIP. The first minor frame in each major frame shall present time code in place of the housekeeping telemetry. The housekeeping telemetry words shall be distributed in the following major frame.

	3.2-sec Digital B Subcom Word*	32-sec Analog Subcom Word	16-sec Analog Subcom Word*	1-sec Analog Subcom Word
Type of data in each word in one minor frame	Eight bilevel status channels	One digitized analog channel	One digitized analog channel	One digitized analog channel
Repetition period of subcom, minor frames	32	320	160	10
Format required to be controlled by ROM within TIP	No	No	Yes	Yes

* There will be two words for each of these subcoms.

The source of data for one word in the l-second subcom shall be controlled in response to a command, and its identity shall be presented in the dwell address identification word.

- Dwell Mode Format -- One minor frame of dwell mode format shall contain a synchronization code, mode status, major frame and minor frame identification, dwell address identification, spacecraft ID, command verification status, command verification, housekeeping telemetry, DAU status and single digitized analog housekeeping telemetry channel. The single channel shall be presented in all the remaining words. It shall be chosen in response to a command, and its identity shall be presented in the dwell mode address identification word. The first minor frame of each major frame shall present time of day data, as in mission mode format.

- CPU Dump Mode Format -- This format shall be identical with the mission mode format, except that the words assigned to CPU telemetry shall all be dedicated to either CPU-A or CPU-B. The choice shall be in response to a command.

- Payload Sensor Inputs (Digital-A) -- TIP shall provide sixteen data input interfaces with payload sensors.

- CPU Data Inputs -- TIP shall provide two input interfaces with the CIU for CPU telemetry data.

- Housekeeping Telemetry Inputs -- TIP shall provide the following input interfaces for housekeeping telemetry.

3.2 second Digital-B subcom-1	352 one-bit digital inputs
3.2 second Digital-B subcom-2	256 one-bit digital inputs
32 second analog subcom	256 analog inputs
1 second analog subcom and 16 second analog subcom-1	256 analog inputs
16 second analog subcom-2	128 analog inputs

The analog overall conversion accuracy shall be nominally better than 1 percent with a worst-case error not greater than 2 percent of full scale.

- Time of Day Counter -- TIP shall provide a 40-bit time of day counter. The count granularity shall be one millisecond. The counter shall be capable of being preset by a command and capable of being made synchronous with the CPU software clock. TIP shall insert the time code into the TIP format as well as outputting the time code to MIRP. Transfer of time code to MIRP shall be once per millisecond.

- Timing -- TIP shall receive clocks from the XSU and synchronization from the CIU.

- Synchronization Outputs -- TIP shall provide l-sec sync, 32-sec (major frame) sync, 128-sec sync, and 256-sec sync, digital-A clock, and digital-A enable signals to payload sensors as required by the Unique Instrument Interface Specifications. The TIP design shall provide one set each of these outputs for up to 16 instruments.

- Data Outputs -- TIP shall provide the following data outputs:

  - Boost Mode -- TIP shall output the boost mode format to the XSU where it will be routed to the S-band transmitter, the DTRs, or the SSRs. The TIP shall provide continuous real-time data for the SCF compatible S-band beacon. No other outputs are required in boost mode.

  - Mission, Dwell, and CPU Dump Modes -- TIP shall output the formats corresponding to each mode to the beacon transmitters, MIRP, and XSU. The TIP shall provide continuous real-time data for the SCF compatible S-band beacon. Outputs to the beacon transmitters shall be split phase.

  - Time Code -- TIP shall output the 40-bit time-of-day code to MIRP.

  - CIU Data -- The TIP shall output the telemetry subcom portion of the orbit mode format to a standard CIU buffer with associated major and minor frame syncs where it can be available to the CPU.

- Redundancy -- The TIP shall consist of two redundant units. Any one unit can be powered at a given time. The input gating shall be designed so that, if an individual digital B or analog input gate fails, the failure shall be confined to a maximum of eight channels. A digital A channel failure shall not contaminate any other digital A channel.

f. MIRP -- MIRP shall process all AVHRR data. It shall format AVHRR data, add synchronization, identification, time code, AMSUor MHS data, and a TIP output data stream as appropriate, and shall produce composite formats for simultaneous output to the XSU and to the real-time VHF and S-band transmitters. MIRP processing of AVHRR data shall consist of formatting, resolution reduction, and geometric correction as required by this specification to match the needs of different user classes.

- Modes -- MIRP modes shall all be selected by command signals from the CIU. All output formats shall be provided simultaneously in all modes except off.

  - Normal Mode -- MIRP shall issue sampling commands to the AVHRR with a fixed phase relationship to the line-synchronization pulses from the AVHRR. MIRP shall generate frame synchronization from internal timing and shall rephase frame synchronization automatically to the line synchronization pulse if the phase displacement exceeds a preset value.

  - Override Resync Logic Mode -- In this mode, MIRP shall operate as in normal mode, except that the automatic rephasing shall be deleted.

  - Backup Mode -- In backup mode, MIRP shall issue sampling commands to AVHRR and shall generate frame synchronization in phase with internal timing only.

  - Test Mode -- Two test modes shall be provided:

    (1) MIRP shall replace the AVHRR parallel input data with a PN code test signal. The code clock rate shall be equal to the data sampling pulse rate. This shall be a commandable function.

    (2) MIRP shall replace the GAC, HRPT, and LAC output data with a PN code clocked at the bit rate of the replaced signal. This shall be a commandable function.

  - Off

- Output Formats -- MIRP shall produce the following formats simultaneously: high resolution picture transmission (HRPT), local area coverage (LAC), global area coverage (GAC), and automatic picture transmission (APT). The major characteristics of the formats are summarized in Table 12.

- AVHRR-MIRP Interface -- The AVHRR-MIRP interface shall provide data sampling commands by MIRP, digitized AVHRR data from AVHRR, and line synchronization pulses from AVHRR. In normal mode and in override resync logic mode, MIRP shall issue sampling commands with a fixed phase relationship to the line synchronization pulses from AVHRR. In backup mode, MIRP shall issue sampling commands in phase with MIRP internal timing. Data sampling logic control will be provided to produce cropped science data and to cause insertion of AVHRR calibration and

Table 12 MIRP Output Formats

	HRPT and LAC	GAC	APT
Form of Data	Serial Digital Bit Stream 10-bit words, most significant bit first		Analog 2.4 Khz Subcarrier
Frame Rate	6/sec	2/sec	2/sec
Word Rate	66.54 K words/sec	6654 word/sec prior to conversion	4160 words/sec
Number of AVHRR Channels Included	5	5	2
Words of Eath Scan per Frame per Channel	2048	409	916 prior to conversion
Processing of AVHRR Data	Formatting and cropping	Resolution reduction, formatting, and cropping	Resolution reduction and correction for geometric distortion, formatting, and cropping
Other Data than AVHRR	TIP; time of day jitter measurement, AMSU	TIP; time of day jitter measurement, AMSU	Minute Marks

housekeeping data into the output data streams, at event times specified in the AVHRR interface document. A unique channel ID wedge shall be provided for each of the 6 AVHRR channels. The MIRP shall provide AVHRR channel 3a/3b indication to the HRPT, GAC, and LAC data streams in ID word #7, bit 10.

- MIRP/AIP Interface--The AIP shall provide combined AMSU-Al, AMSU-A2, and AMSU-B or MHS data to MIRP in NRZ-L format at a 8.32-kbps rate.

- Frame Synchronization--In normal mode, override resync logic mode, and backup mode, MIRP shall generate frame synchronization in phase with MIRP internal timing. In all three modes, MIRP shall measure the time difference between the frame synchronization and the line synchronization pulse from AVHRR. MIRP shall format this time difference into a data word and insert it in the frame header of the HRPT, LAC, and GAC formats. In normal mode, only MIRP shall rephase the frame synchronization automatically to the line synchronization pulse if the above time difference exceeds a preset value.

- APT Channel Selection -- Any two AVHRR channels shall be presented in the APT format. The choice shall be in response to a command.

- Time of Day Counter -- MIRP shall accept time of day counter data from TIP and format it into the header of the HRPT, LAC, and GAC formats.

- GAC Processing -- MIRP shall use one out of every three AVHRR scans for GAC data. MIRP shall compute an average value for four adjacent samples and skip one sample of each channel of AVHRR data across each AVHRR scan line used. It shall insert these averages into the GAC format.

- APT Processing

  - Resolution Reduction and Geometric Distortion Correction -- MIRP shall use one out of every three AVHRR scans for APT data. MIRP shall perform the following schedule of averaging for each of two (selectable by command) channels:

    Use first 121 samples to produce 121 output samples.

    Next 93 samples compute (A+B)/2, (B+C)/2, (D+E)/2, and so forth to produce 62 output samples.

    Next 166 samples compute (A+B)/2, (C+D)/2, and so forth to produce 83 output samples.

    Next 330 samples compute (A+B)/2, (D+E)/2, and so forth to produce 110 output samples.

    Next 628 samples compute (A+B+C+D)/4, (E+F+G+H)/4, and so forth to produce 157 output samples.

    Next 330 samples compute (A+B/2), (D+E)/2, and so forth to produce 110 output samples.

    Next 166 samples compute (A+B)/2, (C+D)/2, and so forth to produce 83 output samples.

    Next 93 samples compute (A+B)/2, (B+C)/2, (D+E)/2, and so forth to produce 62 output samples.

    Last 121 samples use as is, to produce 121 output samples.

    A total of 2048 AVHRR samples shall be used to produce a total of 909 output samples.

  - Conversion of Modulated Subcarrier -- The foregoing output samples for one channel shall be formatted together with synchronization, space scan, and AVHRR telemetry data and shall be digital-to-analog converted with 8-bit granularity at 4160 conversions/second. This shall be repeated for the other channel. The analog signal shall be passed through a third-order transitional Butterworth-Thomson filter with a 2400-Hz 3-dB bandwidth. The filtered signal shall amplitude modulate a 2.4-kHz subcarrier.

- Minute Marker -- MIRP shall provide a one-minute counter to mark in the APT format. The counter shall be capable of being reset by a command.

- Timing -- MIRP shall accept clocks from the XSU, line synchronization from the AVHRR, and TIP minor frame synchronization from TIP.

- Data Outputs -- MIRP shall output HRPT, GAC, and LAC formats to the XSU. MIRP shall output APT format as a modulated subcarrier to each real-time VHF transmitter. All but the initial synchronization words and the auxiliary synchronization words of the GAC and LAC outputs shall be randomized to preclude the transmission of repetitive patterns of continuous ones and zeros.

g. XSU -- The XSU shall provide switched data and clock interfaces with TIP, MIRP, AIP, DTRs, SSRs, and the four S-band transmitters in response to single word commands produced by software within the CPU and delivered by the CIU. It shall distribute timing to the data-handling subsystem and to the payload sensors and shall provide XSU state information to the TIP.

- Modes

  - Recorder Input Modes -- The XSU shall be capable of transferring the data from any of the following sources to the input interface of the DTR (or SSR): either of two identical outputs from TIP/AIP, MIRP LAC data, and MIRP GAC data. Any data stream listed previously shall be capable of being transferred to more than one DTR (or SSR) simultaneously. The bit formats of data to the DTR shall be as specified in IS-2280354 (or UIS-20082617).

  - Meteorological S-band Transmitter Input Modes -- The XSU shall be capable of transferring the playback data from any DTR or (SSR) to the input interface of any S-band transmitter. Also, it shall be capable of transferring data from one TIP or one AIP interface to one S-band transmitter (either boost mode or orbit mode). The bit formats of playback data from the DTRs (or SSRs) shall be as specified in IS-2280354 (or UIS-20082617). The XSU shall preserve these formats in transferring the data to the S-band transmitters. The bit formats of HRPT and real-time TIP/AMSU data to the S-band transmitters shall be split phase.

  - Power Switching Modes--The power switching modes shall be:

    - Side A on, Side B off
    - Side B on, Side A off

- Clocks -- XSU shall accept the master clock from the CIU. It shall distribute clocks to the data-handling subsystem as required and to the payload sensors, including the following:

  - 1.248 MHz to the payload sensors and the AIP
  - 0.9984 MHz to TIP, MIRP
  - 665.4 kHz to DTRs or SSRs for recording LAC data
  - 66.54 kHz to DTRs or SSRs for recording GAC data
  - 8320 Hz to DTRs or SSRs for recording TIP orbit mode data
  - 16.64 kHz to DTRs or SSRs for recording TIP boost and AIP mode data
  - 2.6616 and 1.3308 MHz to DTRs or SSRs for playback of LAC and GAC

- 332.7 kHz to DTRs or SSRs for playback of TIP and AMSU orbit and boost mode data

All clocks shall be coherently derived from the master input from the CIU.

h. AIP -- The AIP shall provide the interface between the three AMSU instruments (AMSU-Al, AMSU-A2, and AMSU-B) and the other spacecraft data handling units.

- Instrument Interfaces shall be as specified in the AMSU UIISs (Table 1).
- The AIP shall accept a data stream from the TIP in all modes.
- Output data
  - The AIP shall output all AMSU data to the MIRP for inclusion in the LAC, GAC, and HRPT data streams.
  - The AIP shall output a combined AMSU/TIP data stream at 16.64 kbs to the XSU for either recording or direct transmission over an S-band transmitter.
- No single point component failure in the AIP shall preclude transmission of AMSU data.

i. DTR -- The DTR will provide recording and playback of LAC, GAC, and TIP data. The DTRs are GFE. Each DTR consists of one electronics assembly and two tape drive assemblies. There are five DTRs in the data-handling subsystem.

j. SSR -- The SSR will provide recording and playback of LAC, GAC, TIP, and AIP data. The SSRs are GFE. Each SSR replaces one or two DTRs. There are three SSRs in the NN' data-handling subsystem. The SSRs are also compatible with the KLM data-handling system.

3.6.1.8 Reaction Control and Propulsion Subsystem

a. Function

NOAA-K,L,M

The reaction control and propulsion subsystem (RCPS) shall provide the required means to achieve the spacecraft/launch vehicle separation as described in paragraph 3.6.1.1b. Thereafter, the RCS shall provide a coast pitch maneuver, attitude control during coast and during the main propulsion burn, a final delta V trim, and attitude control to reduce body rates for array deployment and coast. The operational life of the hydrazine subsystem shall extend until closure of the tank outlet isolation valves after the final velocity trim and prior to the 90° yaw maneuvers. In the event that the ascent phase of the mission does not require use of the entire nitrogen load, The nitrogen subsystem must be capable of using remaining nitrogen for momentum unloading during orbit mode.

The hydrazine system shall use monopropellant hydrazine in accordance with MIL-P-26536, and the nitrogen system shall use nitrogen pressurant in accordance with MIL-P-27401.

NOAA-N and N-prime

The reaction control subsystem (RCS) shall provide attitude control to reduce body rates for array deployment. The nitrogen subsystem shall be capable of using remaining nitrogen for momentum unloading during orbit mode.

b. Configuration -- Components of the reaction control and propulsion subsystem shall include high pressure nitrogen gas tanks, a pressure regulator, a relief valve (with thrust neutralized vent), hydrazine tanks (NOAA-K,L,M only), thrusters, manifolds, filters, pressure transducers, and fill and drain valves as required, as shown in Figure 22a and Figure 22b, and mounted on the aft end of the reaction control equipment support structure. The configuration shall include isolation valves positioned so as to provide isolation between the N_2H_4 tanks and N_2H_4 thrusters.

c. Thrust Levels -- The reaction control subsystem shall provide thrust levels consistent with the ascent control requirements.

d. Thrust Capacity -- The reaction control subsystem shall provide a total impulse sufficient to perform the functions described in paragraph 3.6.1.8a.

e. Tank Capacity -- Capacity of the propellant tanks shall be sufficient to meet the requirements of paragraph 3.6.1.8d.

f. Propellant -- Hydrazine

NOAA-K,L,M

None of the propellant shall be allowed to freeze during the period of required usage. Following, the delta V trim and 90-degree yaw maneuver (see Figure 6 and paragraph 3.1.6), the isolation valves shall be closed and the hydrazine pressure-relieved between the two isolation valves and the four hydrazine thrusters. The hydrazine remaining in the two propellant tanks and tank outlet lines up to and including the isolation valves shall be maintained in the liquid phase at not less than 9°C for the remainder of the mission by an active thermally controlled heater system.

The maximum temperatures of the reaction control equipment subsystem shall be maintained well below those that would cause burst or leak pressures in components. Heat soakback from the apogee kick stage burn and hydrazine engine burns must be considered as well as heat inputs from the Sun, the shroud, and aerothermal heating. The degradation of thermal properties noted during previous NOAA and DMSP flights shall be considered in the design

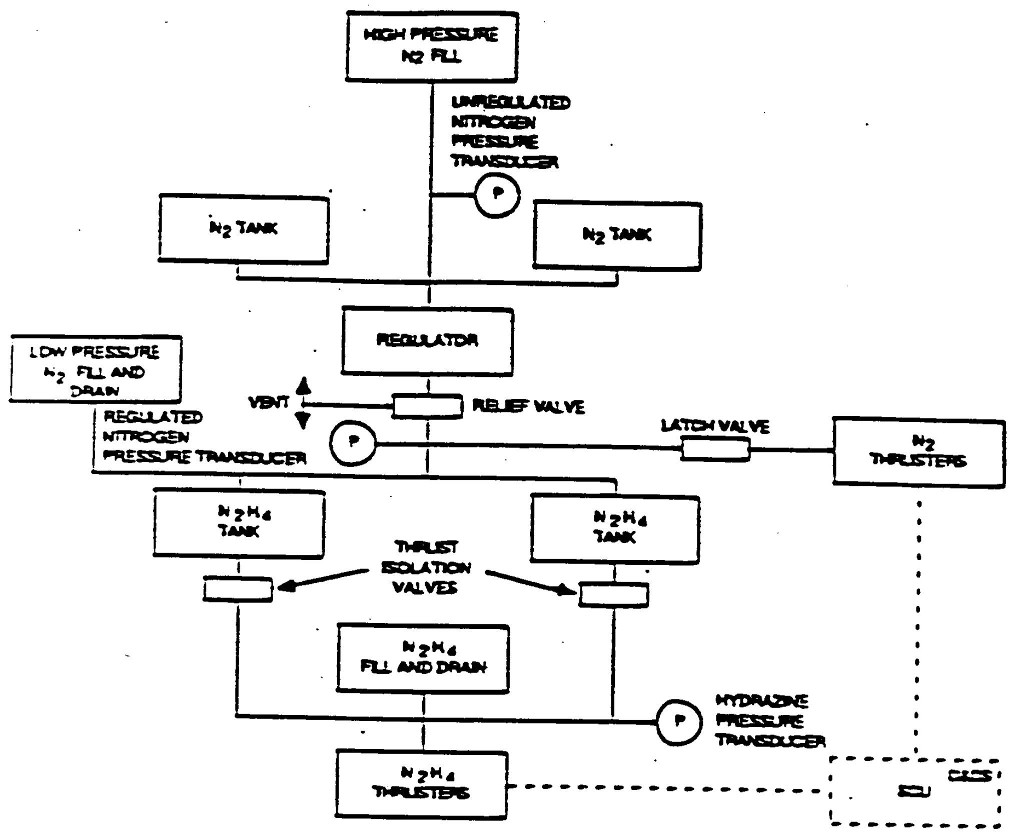

Figure 22a. Reaction Control Subsystem NOAA-K,L, and M

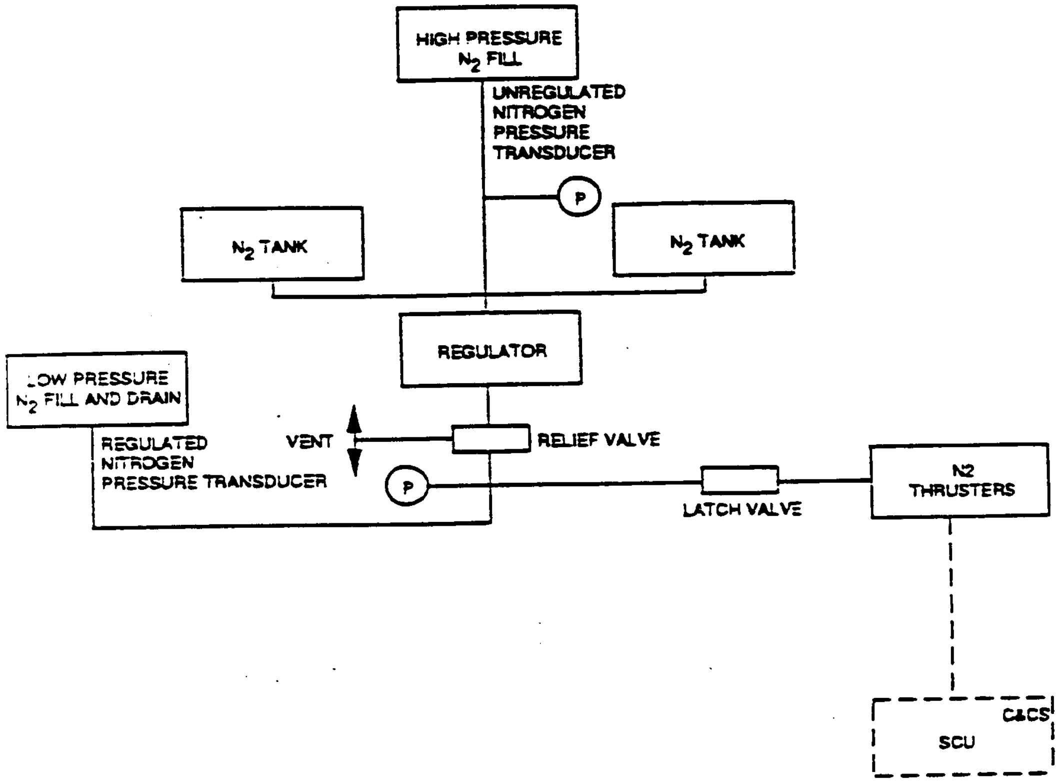

Figure 22b. Reaction Control Subsystem NOAA-N and N-prime

of the RCE thermal control. The subsystem shall meet its leak rate specification for a minimum period of 2 years after the hydrazine thrusters are disabled so as not to cause attitude perturbations or cause contamination of any sensor.

NOAA-N and N-prime

Section not applicable.

g. Thruster Orientation -- All thrusters shall be oriented to avoid causing excessive heating, contamination, or other damage to any satellite unit or surface.

h. RCS Control

NOAA-K,L,M

Attitude determination data shall be provided by the ADACS subsystem and presented to the CPU. Guidance control software located in the CPU shall perform firing control of the RCS thrusters and provide for isolation of the hydrazine thrusters and depletion of the hydrazine in the system from the isolation valves to the hydrazine thrusters.

NOAA-N and N-prime

Attitude data shall be provided by the ADACS and presented to the CPU. Rates control software located in the CPU shall perform firing control of the RCS thrusters.

3.6.1.9 Apogee Kick Stage

NOAA-K, L, and M

The apogee kick stage provides sufficient delta V during ascent so that in conjunction with the TITAN II and N_2H_4 subsystem the satellite achieves the specified orbit.

NOAA-N and N-prime

Section not applicable.

3.6.1.10 Instrument Payload Functional Characteristics

The instrument payload is GFE (Table 2). Refer to the appropriate interface documents for the interface characteristics of the instrument payload (Table 1).

3.6.2 Support Equipment Functional Characteristics

3.6.2.1 Mechanical Equipment

The mechanical support equipment shall include the following items:

a. Factory General-Purpose Equipment -- This equipment shall be that needed to support the satellite during factory assembly and integration. It shall provide access to required satellite locations and shall be used for general handling, assembly, installation, alignment, weighing, balancing, and maintenance. Included shall be general and satellite peculiar equipment, workstands, handling dollies, handling slings, hoisting equipment, fixtures, and adapters.

b. Factory Test Equipment -- This equipment shall be that required to support satellite and subsystem or component development, qualification, and acceptance testing. It shall perform transporting, access, general handling, maintenance, servicing, and testing functions at all test locations.

c. Factory-to-Site Transportation Equipment -- This equipment shall be that necessary to house and protect the satellite during transport from the factory to test and launch sites. It shall be designed for truck and air transportation and shall include necessary means for handling and tiedown. The contractors shall specify the maximum environmental conditions to which the spacecraft shipping container can be subjected without adversely affecting the spacecraft inside. Instrumentation shall be provided to monitor or record all controlled environments continuously. The contractors shall provide instruments to monitor the temperature and humidity inside the spacecraft shipping container. The government will provide vibration-monitoring equipment for the spacecraft shipping container.

3.6.2.2 Electrical Equipment

The electrical support equipment shall meet the requirements of 2285033 and Appendix A and shall be modified to meet the requirements of 2295960. This equipment, in conjunction with standard test equipment, shall provide the capability to verify the integrity of all launch critical elements of the satellite, provide power, provide control functions, monitor performance, and provide failure indication at the component, subsystem, and system level. The electrical equipment shall not cause satellite component or system failures and shall also provide permanent records of test data. The design of electrical equipment shall incorporate means to protect it and the satellite from damage as a result of power source interruption. Electrical support equipment shall include the following:

a. Factory Test Equipment -- This equipment shall be required to support satellite development, qualification, and acceptance testing at all test locations, and shall perform the following:

- Conduct the required subsystem and system level tests through the use of automated test equipment
- Collect resulting test data and provide computational and analytical capability
- Provide continuous monitoring of all predictable parameters for which unpredictable behavior should be analyzed
- Provide automatic collection of data, produce history data, and retrieve and provide viewing capability for these data
- Provide a controlled operator interactive access to the test system control, observations of the test progress, system status, and test data outputs
- Provide a mechanism to determine system faults and aid in the protection of the spacecraft

b. Launch-Site Equipment -- This equipment shall be that necessary to support launch-site, receiving/inspection of the satellite, testing to verify satellite buildup and integration, interface checkout, and system integrated test before movement to the launch pad.

c. Launch-Pad Equipment -- This equipment shall be that needed to support prelaunch electrical checkout of the satellite at the launch pad.

3.6.2.3 Reaction Control System Support Equipment

This support equipment shall include all equipment used to support the manufacture, development, qualification, test and launch of the NOAA-K, L, M, N, and N-prime spacecraft RCE. This equipment shall be categorized in two general classes.

- Class I -- Equipment dedicated to the NOAA-K, L, M, N, and N-prime.

- Class II -- Equipment that is considered to be standard factory or launch-site equipment that is also used to support other satellite contractor programs.

The functional characteristics of the equipment are as follows:

a. Factory Equipment -- Factory equipment shall be that which is needed to support the assembly, development, and of the RCE at the system and spacecraft level. The equipment shall also be used to decontaminate, maintain, and test service launch-site equipment at the factory. The equipment shall support the parts fabrication, component level test, cleaning, assembly, brazing and welding, pressurization, leak test of components and the system, electrical test, thermal test, and integration test of the RCE and the spacecraft. Included are all fixtures, adapters, precision cleaning fixtures, clean room fixtures, workstands, RCE handling dollies, RCE slings, pressurization panels (including pressure hoses, manifolds, and fittings), leak test equipment (including enclosures).

b. Launch-Site Equipment -- This equipment shall be used during launch-site operations to:

- Pressurize and leak-test the RCE
- Prepare the propellant loading cart and load the RCE with the prescribed weight of hydrazine propellant (NOAA-K,L,M only)
- Pressurize the RCE to flight pressure with the prescribed quantity of nitrogen pressurant

and if necessary to:

- Vent the pressurant nitrogen from the RCE
- Offload the hydrazine propellant from the RCE, (NOAA-K,L,M only)
- Troubleshoot the RCE

The launch-site equipment will include high and low GN2 pressure panels and leakage-measuring equipment. The contractor shall further provide hydrazine detection equipment, off-load equipment (NOAA-K,L,M only), drums, flex lines, fittings, valves, and safety equipment. The previously mentioned launch-site equipment is necessary to support the launch activities for the NOAA-K, L, M, N, and N-prime spacecraft.

The hydrazine loading cart will be provided as GFE, but operation and maintenance of the cart shall be provided by the contractor. (NOAA-K,L,M only)

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3.6.3 Spacecraft Software Subsystem

3.6.3.1 Unified Load Package

The unified load package (ULP) shall perform functions including, but not limited to, the following:

- Provide vehicle attitude determination from liftoff (NOAA-K,L,M,) or from Earth acquisition (NOAA-N and N-prime) to handover to orbit mode operation.
- Provide a prelaunch test capability of the navigation system. (NOAA-K,L,M only)
- Provide a prelaunch test capability of system alignment. (NOAA-K,L,M only)
- Provide a prelaunch test capability of RCE operation.
- Provide a prelaunch simulated flight test capability.
- Interface with the CIU.
- Provide CPU telemetry to TIP.
- Accept and issue vehicle discretes including, but not limited to, stage separation, solid motor ignition (NOAA-K,L,M only), and antenna and solar array deployment.
- Provide ascent guidance function (NOAA-K,L,M only) and control signals to RCE for control from booster/spacecraft stage separation to orbit mode handover.
- Accept stored and real-time commands from the CIU.
- Transform valid real-time, stored and internal command into required action (i.e., control information to CIU).
- Generate a positive command verification signal for transfer to TIP.
- Provide a capability to dump CPU memory via TIP output data.
- Collect and output preprogrammed memory words for transfer to TIP in accordance with a preprogrammed schedule.
- Provide a capability to switch logical control between the two CPUs, including internal diagnostics and operating mode control.
- Accept outputs from ESA, IMU, SSD, and RWAs and, using uploaded stored ephemeris data, produce control signals to RWAs and momentum coils to control spacecraft attitude and momentum as required by paragraph 3.6.1.3e. In addition, provide capability for automatic unloading of momentum wheel speeds by actuation of GN_2 thrusters. Enable/disable thresholds shall be adjustable for each axis. Provide for initial acquisition assist in the event of a primary wheel failure by uplink commands to the flight computer. Thruster pulse repetition shall be adjustable for each of the three axes by uplink command.
- Using input from ADE and stored ephemeris data, produce drive signals to ADE to maintain proper orientation between the solar array and the solar vector.
- Perform upload message verification checks.
- Perform memory section check calculations and report upon command.
- Provide power management software having failure protection capabilities that include battery charge level monitoring, battery temperature and pack temperature gradient monitoring, battery voltage monitoring, and Sun loss monitoring with capability to automatically switch to a safe operational mode. There shall be an onboard filtering capability for programmable analog parameters on l-second telemetry subcom.
- The ULP shall provide fully symmetrical, redundant functional capabilities in both CPUs: one acting as "control" and the other as "standby". Under ground control, either computer can be designated "control" or "standby."

3.6.3.2 Extended Power-on Processor Software

The extended power-on processor software (XPOPS) package shall provide the emergency means for loading data into a CPU read/write memory regardless of either the existing contents of read/write memory or the execution mode of the CPU. The requirements for this software are contained in PS-2303078.

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3.6.4 Ground-Support Software System

3.6.4.1 Operating Systems

The operating systems software within the Ground Support Software System (GSSS) shall be capable of executing the applications software required to support the Integration and Test (I&T) and launch activities of the NOAA-K, L, M, N, and N-prime spacecraft utilizing (1) 32-bit addressing the A, B and C computers of each ATNAGE, the Simulator and the Off-line systems and (2) 16-bit addressing in the A computer of each ATNAGE and the Offline system. The dual 32-bit or 16-bit addressing capability within the computers shall be retained until the applications software supporting the NOAA-I and NOAA-J spacecraft activities have been converted to operate on a 32-bit system or until both spacecraft have been launched, at which time the 16-bit capability may be deleted. The dual 32-bit or 16-bit addressing capability shall be maintained in the Off-line system as long as any required data analysis software operates exclusively in 16-bit mode.

3.6.4.2 Applications Software

The existing and additional applications and utilities software that is required to support the NOAA-K, L, M, N, and N-prime I&T and launch activities shall be modified or developed, respectively, to operate on the ATNAGE and ancillary ADP equipment within the GSSS (in 16-bit or 32-bit mode on the Offline system and in the 32-bit mode on all other systems within the GSSS) and shall satisfy the requirements of Appendix A herein.

3.6.5 Launch Vehicle Function Characteristics

NOAA-K, L, and M

Refer to the Titan II ICD to the ATN spacecraft (NOAA-KLM), ICD-TII-25004, for the functional characteristics of the launch vehicle system.

NOAA-N and N-prime

Refer to the Delta II National Oceanic and Atmospheric Administration – N (NOAA-N) Mission Specification MDC 00H0072 October 2001 to the ATN spacecraft (NOAA-N), for the functional characteristics of the launch vehicle system. NOAA-N prime TBD. MOD 481 CCR 1891A

3.6.6 CDA Site Hardware and Encryption/Authentication Equipment

The contractor shall provide equipment to interface with the DCA PACS system to provide for encryption and command operation (STDN) for NOAA-K, -L,-M, -N, and -N-prime satellites. This equipment shall utilize NSA KIT-23 EMS technology. A redundant set of equipment shall be provided.

3.6.6.1 Encrypting Authenticating Equipment

The encrypting/ authenticating equipment called the command encryption module (CEM) shall interface with the CDA PACS Host computer (VAX) to provide an encrypted digital signal. These digital signals shall interface with the command generator (STDN) via an RS232C protocol. Normal prepass or pass activities should not require operator intervention including switching between satellites. The CEM equipment shall contain self-test modes.

3.6.6.2 Command Generator Equipment

The contractor shall supply a STDN Command Generator to interface with the output of the VAX (PACS) via a RS232C interface. This equipment shall format the digital signal input to STDN format and modulate a 16 KHz subcarrier for output to the RF Signal Generator also provided as part of the CDA site equipment.

3.6.6.2.1 Manual Command Generator

As part of the Command Generator equipment, a manual input capability shall be provided to permit the sending of STDN formatted commands to the orbiting satellites in the clear mode. Equipment shall be operator controlled and shall be capable to prestore and display ten commands that have been entered manually. A manual keyboard shall be provided on the front panel to enter commands in hexidecimal opcode.

3.6.6.3 CDA Site Command Uplink Antenna Probe Verification Function

As part of the CDA site equipment, the contractor shall provide equipment to verify the command uplink transmission signal utilizing the CDA antenna probe.

A phase modulation receiver operating at 65 MHz shall be provided to demodulate the uplink data from the CDA antenna probe. This 16 KHz subcarrier data shall be sent to the command generator assembly to be demodulated to 2 Kbs NRZ-M baseband data to provide Bit Error Status as part of the command verification system. This data will be available for interface to the PACS (NAGE) equipment.

3.6.6.4 Spacecraft Ground Test Equipment (NAGE)

The spacecraft ground test equipment (NAGE) shall utilize two identical versions of the equipment.

3.6.6.5 CDA Site Interface Specification

The contractor shall, with the assistance of NASA, develop interface specifications for the above equipment.

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3.7 PRODUCT ASSURANCE REQUIREMENTS

The contractor shall develop and implement reliability, quality assurance, system safety and configuration management systems that meet the requirements of this specification, the Statement of Work, and the Performance Assurance Requirements (S-480-26.1); however, it is intended that the contractor use his programs as a base and modify them as required to meet the NASA requirements.

4 ENVIRONMENTAL TEST PROGRAM

4.1 GENERAL

The test program in general will apply to all hardware divided into four general categories:

- Component tests
- Design verification tests
- Spacecraft bus evaluation tests
- Satellite flight acceptance tests

4.1.1 Preparation for Test Program

The satellite and components submitted for testing shall have all protective covering, potting, thermal insulation, shock mounting, or any other means of attenuating environmental rigors expected in flight. Any appendage or other attachment that will render the item being tested more vulnerable shall be included or simulated. Flight models submitted for acceptance testing shall be replicas except for approved ECNs of the qualified protoflight. A component (where feasible and agreed to by the contractor and the government) and a satellite undergoing environmental exposure in either a protoflight or acceptance test program shall be in the operating mode expected of it during the launch or orbital condition being simulated by the environmental exposure. Any changes to the test plan shall be submitted 90 days before test for GSFC approval.

4.1.2 Detailed Test Procedure

A detailed step-by-step test procedure shall be prepared before each environmental exposure or test. This detailed test document must be available for review by the cognizant GSFC technical personnel 15 days before the test or exposure. Any changes to the detailed test procedure must be available for review by the cognizant GSFC representative. A copy of the detailed test procedure with all changes shall be maintained by the contractor, and a log will be maintained of each change with the date of such change.

4.1.3 Test Facilities

4.1.3.1 General

The apparatus used in conducting the test shall be capable of producing and maintaining the test conditions required.

4.1.3.2 Standard Conditions for Test Area

Laboratory conditions for conducting specimen operational checkout before or after an environmental exposure shall be as indicated, unless the specimen is sealed, protected, or otherwise functionally insensitive to variation in temperature and humidity. In those cases, checkout at room ambient conditions shall be acceptable.

- Temperature: 24° ±5°C
- Relative Humidity: 30 to 55% (25 to 30% if wrist stats are worn)
- Barometric Pressure: Room ambient with performance data corrected to 760 torr (29.98 inches of mercury), if specified in the particular detailed test procedure.

4.1.4 Measurement and Test Equipment

All measurements shall be made with instruments that are appropriate for the category involved and for the environmental conditions concerned. All standard commercial inspection, measuring, and test equipment shall be calibrated at scheduled intervals against certified standards that have known valid relationships to national standards. Records shall be maintained indicating the date of last calibration and due date. The due date or other identification attesting the due date of the next calibration shall be displayed on each item of inspection, measuring, and test equipment. Procedures shall be generated by the responsible personnel that provide the means for periodic operational checks to be performed before use of each inspection, measurement, or test equipment.

4.1.4.1 Tolerances

Unless otherwise specified by the applicable test procedure, the maximum allowable tolerances for test conditions shall conform to the following requirements.

- Temperature (sensor accuracy): +3°C (on temperature control sensors)
- Relative Humidity: +0, -5% R.H.
- Vibration Amplitude
  - Sinusoidal: ±10%
  - Random - overall rms level: ±10%
  - PSD: ±3% dB
- Vibration Frequency: ±2% or 1 Hz (whichever is greater)
- Weight
  - Spacecraft: ±.25%
  - Subs/exp: ±0.1%
- Additional tolerances shall be as specified.

4.1.5 Section Reserved

4.1.6 Installation Check

Following spacecraft or component installation in a test apparatus and before exposure to the test environment, the spacecraft or component shall be operated electrically to ensure that malfunction or damage was not caused by a faulty installation procedure or handling.

4.1.7 Criteria for Unsatisfactory Performance or Construction

Any detectable change that can be interpreted as leading to failure or harmful degradation within the expected operational life or any change in the operation as defined in the applicable performance specification shall be interpreted as a discrepancy or failure. This determination will be made jointly by NASA and the contractor.

4.1.8 Integration and Test

Upon recognition of a failure or discrepancy during a test, the test director will discontinue the test if there is any risk to any component. If there is a risk to any component, the test will be continued with the approval of both the contractor's program manager or his authorized representative and the GSFC Tiros spacecraft manager or his authorized representative, who will be notified at the first opportunity after the failure or discrepancy occurs.

A Test Review Board (TRB) shall be convened within 24 hours to make a final determination of test continuance or discontinuance. The TRB will also establish the level of retest required for in-line failures or failures with retroactive effect.

4.1.8.1 Malfunction Reporting

Malfunctions or failures shall be reported in accordance with the provisions of GSFC S-480-26.1. Reporting shall begin with the first functional test of an assembly or subassembly and continue throughout the operational life of the item as required by the reference document.

4.1.9 Substitution of Components

If a component is operated in excess of design life and wears out or becomes unsuitable for further testing during a system acceptance test sequence due to a cause other than design deficiencies, a different component may be substituted. However, if the substitution affects the significance of results of the test sequence during which the part failed, the test sequence and any previously completed procedures that are affected shall also be repeated.

4.1.10 Transportation and Handling

To ensure that environmental conditions resulting from transportation and handling do not exceed the levels imposed by the tests, thereby imposing unnecessary penalty on the design of the spacecraft, the shipping and handling environment shall be controlled and monitored for specified modes of transportation, handling, and by the use of properly designed shipping containers.

4.2 COMPONENT TESTS

4.2.1 Protoflight Testing

The first set of new or redesigned components shall be tested to protoflight standards. For vibration, this means that prototype (design qualification) test levels at flight acceptance durations shall be used. For thermal vacuum, the temperature extremes shall be ±10°C over and above those determined to be worst case expected during launch and orbit. The vibration levels and thermal vacuum temperature extremes shall be recommended by the contractor and approved by GSFC.

4.2.1.1 Weight

The component weight shall be determined.

4.2.1.2 Vibration

The subsystem shall be attached to a fixture where the attachment points shall simulate the spacecraft structure with regard to hole pattern, torque, preload, and bolt size. The mounting fixture shall be attached to the vibration equipment so that the component may be vibrated in each of three orthogonal directions, one of which shall be parallel to the thrust axis. The subsystem shall be exposed to random vibration in one axis before proceeding to the next axis. An X-Y plot of the filter voltage output or power spectral density versus frequency is required for each axis. In case of unexpected equipment problems, the use of tape recording the data, especially the control accelerometer signal, is highly recommended. During vibration exposure, the component (where feasible and agreed to by the contractor and the government) shall be operated in its launch mode.

4.2.1.3 Launch Loads

Structurally modified components shall be subjected to a loads test (preferred) or an analysis (option) of the primary structure demonstrating design adequacy. The new or redesigned hardware shall be subjected to a load/acceleration test of 15g longitudinal and 3.75g lateral applied simultaneously (modification 17) or a test equal to 1.25 times the expected flight (limit) load in three axes.

If the analysis option is chosen, then a positive margin must be demonstrated using one of the following criterion:

a. A longitudinal load of 24.0 g's coupled with a lateral load of 6.0 g's (modification 17).

b. Positive strength margins shall be shown to exist at stresses equal to 2.0 times those induced by the limit load on yield and 2.6 times those induced by the limit load on ultimate.

4.2.1.4 Leak

All hermetically sealed units shall be subjected to a leak test to ensure performance to the individual specifications.

4.2.1.5 Thermal Vacuum

Protoflight subsystems shall be exposed to a thermal vacuum test consistent with requirements of the PAR. Where it can be demonstrated that the subsystem is not vacuum sensitive (i.e., nonhigh voltage and sealed), a thermal test may be substituted. The test shall comprise a minimum of eight hot and eight cold cycles and the total duration at the plateaus shall be 32 hours hot and 32 hours cold.

4.2.2 Flight Testing

NOAA-K, L, M, N, and N-prime components shall be tested to flight levels except for new or modified components which will have the first article tested to protoflight levels.

4.2.2.1 Vibration

A three-axis workmanship random vibration test shall be performed according to the applicable Performance Specification or as agreed to between the contractor and the GSFC Tiros Project.

4.2.2.2 Leak

The contents of paragraph 4.2.1.4 shall apply.

4.2.2.3 Thermal Vacuum

The contents of paragraph 4.2.1.5 shall apply except that the temperature extremes shall be those determined to be worst case for launch and orbit.

4.3 NOAA-K, L, M, N, and N-prime PREACCEPTANCE SPACECRAFT TESTING

The spacecraft components and instruments shall receive an initial power and functional and detail electrical test as soon as practical after installation on the spacecraft.

4.3.1 Leak Test

The satellite shall be subject to an RCE leak test prior to flight acceptance testing.

4.4 NOAA-K, L, M, N, and N-prime FLIGHT ACCEPTANCE TESTS

The satellite flight acceptance test program is intended to demonstrate the successful performance of the complete satellite. The levels of test are planned to be equivalent to those expected during launch and orbit. The spacecraft flight acceptance test program shall be comprised of the following paragraphs.

4.4.1 System Electrical Performance Test (SEPET

4.4.1.1 Purpose

The purpose of this test will be to verify electrical performance of all systems during the satellite flight environmental test program. Satisfactory electrical performance in all applicable modes before, during, and after the specified environments shall be required. The test will be based on exercising all redundant functions but not in all combinations of each other. All recorders will be filled at least once during the test. The goal of the test will be to perform all necessary tests in an ascent mode sequence followed by orbit mode tests.

The aliveness test shall be a subset of the SEPET not requiring filling recorders or exercising all modes of operation. The goal is to perform the aliveness test in approximately 4 hours. Thermal vacuum SEPET testing is to be essentially the same as the system electrical performance test in air with modifications necessary for thermal vacuum operation. The thermal vacuum SEPET shall be performed once for each T/V temperature plateau level, except for NOAA-M, which shall only have SEPETs performed at the hot and cold plateau. The spacecraft operational test (SOT) shall be conducted for the remainder of the time. The electromagnetic interference/electromagnetic compatibility (EMI/EMC) test shall be a derivation of the system electrical performance test; however, complete recorder fills will not normally be required. During acoustics, the ascent phase of the system electrical performance test shall be conducted.

4.4.1.2 Times of Performance

The initial SEPET test shall be conducted prior to the beginning of environmental tests to determine whether electrical performance meets the requirements of the satellite specification. The SEPET will be repeated prior to thermal-vacuum testing, at least two times during thermal-vacuum testing (three for NOAA-K and L), prior to acoustic testing, prior to shipment to the launch site, and after arrival at the launch site, to determine whether the test, handling, and transportation environments adversely affected the satellite's performance. MOD 427 CCR 1704

The aliveness test will be performed at the following points in the integration and test program:

a. Following EMI/EMC and before thermal vacuum

b. Prior to acoustics and operational shock

c. Following acoustics and operational shock

d. Prior to shipment to the launch site

e. Following encapsulation in the booster fairing at WTR

f. Following the spacecraft mate to the booster at WTR

4.4.2 Mass Properties, Weight, and Center of Gravity

While nonoperative, the satellite weight and center of gravity shall be measured. Moments and products of inertia of the spacecraft in all three axes shall be calculated for the launch configuration unless otherwise specified.

4.4.3 EMI/EMC

Sufficient emissions and susceptibility tests shall be conducted to determine that the satellite electrical and electronic equipment will operate satisfactorily in orbit. This test will be conducted at the contractor's facility in the orbit mode and in the launch mode.

4.4.3.1 RF Radiation Test

A test to verify proper RF radiation levels in the orbit configuration shall be conducted.

4.4.4 Sinusoidal Vibration

A sinusoidal vibration test, to simulate the expected low frequency transient vibration responses present during the launch ascent phase, shall be conducted on the spacecraft. The vibration test shall be conducted with the spacecraft in its flight configuration. Notching and limiter accelerometers shall be permitted to protect the structure.

For each lateral axis test, the frequency limit shall be chosen to be no more than 10Hz above the second spacecraft bending mode for that axis. For the longitudinal test, the frequency limit shall be no more than 10Hz above the frequency for the space thrust mode. Results of a low level sine sweep shall be used to determine the upper frequency limits.

The spacecraft contractor will develop the sine vibration profile for the NOAA-K, -L, and -M spacecraft. The spacecraft contractor and GSFC will jointly develop the sine vibration profile for the NOAA-N and -N prime spacecraft.

4.4.4.1 Qualification (NOAA-K and N)

The NOAA-K and NOAA-N spacecraft shall be tested to qualification levels. A qualification level test shall generate responses within the structure that are 1.25 times the expected flight levels. Notching and limiter accelerometers shall be permitted in order not to exceed qualification levels as determined from the spacecraft’s Verification Coupled Loads Analysis.

4.4.4.2 Acceptance (NOAA-L and M)

The NOAA-L and -M spacecraft shall only be exposed to a single axis test along the longitudinal, or thrust, axis. Notching and limiter accelerometers shall be permitted in order not to exceed flight levels as determined from the Verification Coupled Loads Analysis.

4.4.4.3 Acceptance (NOAA-N-prime)

The NOAA-N prime spacecraft shall be tested to acceptance levels. An acceptance level test shall generate responses within the structure that represent the expected flight levels. Notching and limiter accelerometers shall be permitted in order not to exceed flight levels as determined from the Verification Coupled Loads Analysis.

4.4.5 Acoustics

The satellite in the launch mode, except for fairing, shall be subjected to an acoustic test. The test level and duration shall be as shown in Table 4. For NOAA-K and NOAA-N the test level shall be increased by 3 dB. A system performance test and deployments test shall be performed before and after the acoustic test.

4.4.6 Mechanical/Operational Shock

An operational shock test shall be performed following the sine and acoustic tests. This test will consist of firing the ordinances of the V-band bolt cutter and all deployable appendages and protective cover. After the test, a visual inspection will be performed to verify that the two V-band halves are clear of the separable interface and all appendages and protective covers have performed within specification. For NOAA-K and NOAA-N, the test shall also be performed prior to the sine/acoustic tests. For NOAA-M and NOAA-N' the firing of the V-band bolt cutter ordinance shall not be required. Also for NOAA-M and NOAA-N', shock instrumentation and recording shall only be required in the vicinity of non-shock qualified spacecraft hardware (ie. For SSR accommodation on NOAA-M, pyro shock data shall be recorded for solar array panel and solar array boom deployments).

4.4.6.1 Handling and Transportation

Satellite design should not be influenced by handling and transportation loads. Shocks thus encountered should be less severe than those discussed in the foregoing tests. The dynamic loads transmitted to the satellite shall be limited by appropriate shipping container design with considerations of ground and air vehicle characteristics and specific handling procedures. The reusable shipping container shall be capable of being pressurized with dry nitrogen during shipping and storage. A gauge shall be incorporated to facilitate checking the pressure in the container.

4.4.7 Thermal Balance/Thermal Vacuum

The thermal balance test shall not be performed unless design changes are implemented for NOAA-K, -L, -M, -N, or -N-prime which would significantly affect the present ATN thermal design.

The thermal vacuum test shall be performed to demonstrate the ability of the satellite to perform properly at the extreme and nominal modes of operation required by the mission, while under simulated vacuum ($\rightarrow 1 \times 10^{-5}$ torr) and temperatures at the predicted extremes. For NOAA-K and NOAA-N, a ±5^{o}C test margin shall be added to the predicted temperature extremes.

The thermal vacuum test shall conform to the generalized profile shown in Figure 23. The specific profile shall be contained in the thermal vacuum test procedure to be used for each spacecraft. Only approved materials shall be used in the test. Before the test, all test equipment and materials (cables, hardware, and blankets) shall be baked out. The bake-out shall be done in vacuum at 80°C for at least 48 hours. For the chamber that is used to do the satellite thermal vacuum test, chamber cleanliness shall be established by installing two QCMs that are maintained at -10°C ±2 C. A QCM count change of 0 ±5 counts (Hz/hr) shall be acceptable with all of the chamber's shrouds set to a common temperature equal to the maximum of their specified (hot case) test temperature. Contamination monitoring devices and witness mirrors shall be used in the chamber during the thermal vacuum test.

Preparatory to all thermal vacuum testing, the location, sizes, and desired temperatures of all GFE instrument targets shall be determined. The combined thermal effects of the targets and the chamber shroud and heaters shall be analyzed to result in the desired test temperature limits on all GFE instruments. A single target shall be used as a simulated space background for calibration checks of the Earth sensor assembly (ESA) detectors. The target will be matte black, liquid nitrogen cooled, and cover all four detectors.

During the test, all active and passive louver set-points shall be verified. All heater actuation set-points shall be verified for all heater controllers. Verification of flight temperature telemetry accuracy shall be made by test thermocouples attached reasonably near all accessible flight temperature sensors.

The external pressure integrity of the spacecraft RCS shall be verified under space environmental conditions in conjunction with the thermal vacuum test. The propulsion system shall be pressurized to a condition that is representative of the normal in-orbit pressure using nitrogen gas mixed with approximately 10 percent (by volume) of argon as a trace gas. The argon gas content in the test chamber shall be monitored during the performance of the vacuum test for evidence of leakage from the RCS. If the argon content is higher than the normal background level for

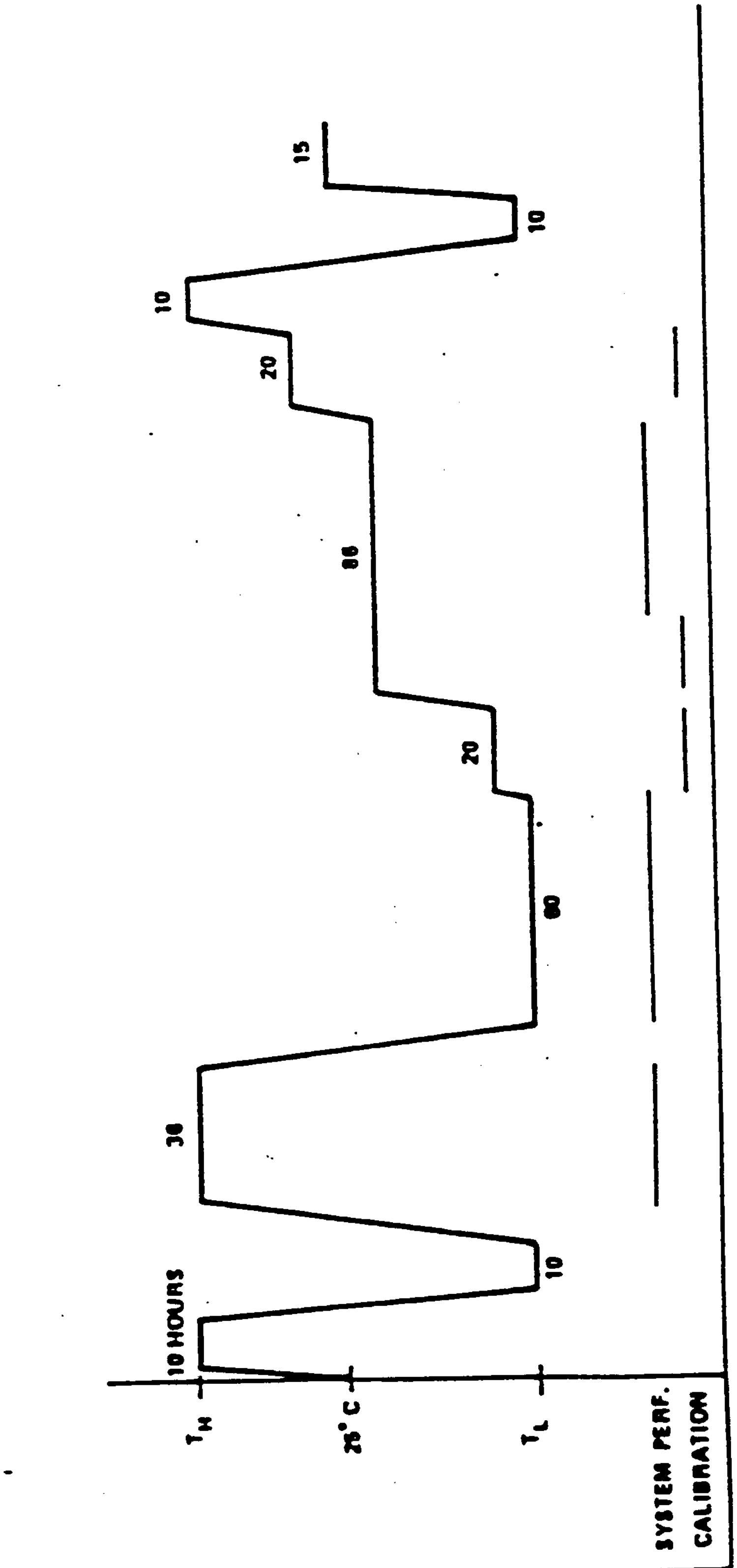

Figure 23. Generalized Thermal Vacuum Profile
(may not be representative of a particular spacecraft)

the chamber, the reason for the condition shall be investigated.

4.4.7.1 RCS Configuration (NOAA-K,L,M only)

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The nominal in-orbit pressure is defined as 1000±50 psia in the nitrogen tanks and system upstream from the regulator and "regulator lockup" pressure in the system down-stream from the regulator. The RCS configuration during the thermal vacuum test shall be as follows:

- Latch valve closed.
- High-pressure fill-drain valve closed with flight cap installed.
- The low-pressure fill valves are closed. The interconnect manifold is removed.
- All thruster valves shall be electrically disabled.
- The thruster nozzles shall be uncapped and open to the chamber. Before the start of the thermal vacuum test, each thruster shall be exercised to verify its operation by gas flow. Nitrogen gas shall be supplied at approximately 30 psia. The thrusters shall then be disabled and the system pressurized to the prescribed thermal vacuum test pressure. The latch valve shall then be closed and the thermal vacuum test performed. At the conclusion of the test, the RCS pressure shall be vented to "pad" pressure and the system returned to its normal storage condition.

4.4.7.2 RCS Configuration (N,N-prime only)

The nominal in-orbit pressure is defined as 1000 ± 50 psia in the nitrogen tanks. The RCS configuration during the thermal vacuum test shall be as follows:

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- High-pressure fill-drain valve closed with flight cap installed.
- All thruster valves shall be electrically disabled.
- The thruster nozzles shall be uncapped and open to the chamber. Before the start of the thermal vacuum test, each thruster shall be exercised to verify its operation by gas flow. Nitrogen gas shall be supplied at approximately 1000 psia. The thrusters shall then be disabled and the system pressurized to the prescribed thermal vacuum test pressure. At the conclusion of the test, the RCS pressure shall be vented to “pad” pressure and the system returned to its normal storage condition

4.4.8 Leak

Subsystems (components) and instruments (sensors) that operate as hermetically sealed units shall be subjected to a leak check or have a capability for monitoring internal pressure. Indirect measurement (e.g., RWA rundown time) may be used instead of direct measurements if supported by analysis.

4.4.8.1 Leak Rate

The permissible RCE leak rate shall be as follows:

a.1 Total System (NOAA-K,L,M) -- The total permissible system leak rate, including external and internal sources, shall not exceed 15 scc/hr of helium with the N_2 thruster latch valve closed, the system pressurized with helium to 1000 psi in the GN_2 supply tanks, and the regulator lockup pressure in the system down-stream of the regulator.

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a.2 Total System (NOAA-N,N’) – The total permissible system leak rate, including external and internal sources, shall not exceed 22 scc/hr of helium with the system pressurized with helium to 1000 psi in the GN_2 supply tanks.

b. External -- In addition to the total system leak rate specified in 4.4.8.la, the external leakage from any one component, tank, and plumbing junctions shall not exceed 100x the background ambient reading of helium when pressurized to 1000 psia. CCR 2422R1 MOD 566

c. Internal -- Internal leakage through any component shall not exceed the values specified in Table 13.

4.4.8.2 Measurement

The leak rate shall be measured with the appropriate leak detection system. Prior to the test, the detection systems shall be calibrated against a standard leak device, and the magnitude of the background of the greater gas in the test chamber shall be determined.

4.4.9 Alignment

An initial alignment test shall be conducted before system mechanical testing begins. Alignment checks should be made after major exposures, such as acoustics and operational-shock testing.

4.4.10 Solar Array Illumination

A solar array illumination test shall be conducted after the postvibration solar array deployment test. The illumination test shall provide a power system end-to-end check and shall verify overall performance.

Table 13 Maximum Allowable Leakage for RCE Components

Component	Test Gas	Test Pressure	Test Temperature	Maximum Allowable S/C Leak Rate
Hydrazine Thruster Engine (valve) (NOAA-K,L,M)	GN_2	100 and 465 psia	Ambient	3.0 scc/hr
N_2 Thruster (NOAA-K,L,M)	GN_2	465 psia	Ambient	3.0 scc/hr
N_2 Thruster (NOAA-NN')	GN_2	1000 psia	Ambient	3.0 scc/hr
Relief Valve (NOAA-K,L,M)	GN_2	575 psia	Ambient	3.0 scc/hr
Regulator (NOAA-K,L,M)	GN_2	Lockup Pressure	Ambient	3.0 scc/hr
Latch Valve (NOAA-K,L,M)	GN_2	0 to 550 psia across valve seat	-29° to 66°C gradient not to exceed 16°C/ minute	3.0 scc/hr
Fill-Drain Valve (NOAA-K,L,M)	GHe	550 psia	Ambient	3.0 scc/hr
Fill-Drain Valve (NOAA-N,N-prime)	GN_2	4500 psia	Ambient	3.0 scc/hr

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Table 14 Environmental Test Matrix

Tests / Components	Mass Properties	Weight and Balance	Structural Loads	Random Vibration	Leak	Thermal Vacuum	Deployment	Operational Shock	Thermal Balance	Alignment	EMI / EMC	Acoustics
New Design Flight Components/Subsystems		X	X *****	X	X ***	X *					X	
Flight Components/Subsystems				X	X ***	X *						
Satellite	X	X	X ****	X	X	X	X	X	X **	X	X	X

* Thermal test may be substituted subject to GSFC approval.
** Not required if significant design changes are not incorporated.
*** Leak testing is required only on hermetically sealed components and subsystems.
**** Will be performed on NOAA-K spacecraft only.
***** Performed only if need indicated by analysis.

4.4.11 Mechanical Deployments

Antenna, solar array panels, solar array boom and cant, MEPED (where applicable), and sunshade deployment tests shall be conducted on a fully configured satellite both before and after the sine and acoustics tests. On NOAA-K and NOAA-N, both the pre- and post-acoustic deployments shall be initiated via pyrotechnics. For NOAA-L, M,& N-prime, only the post-acoustic deployments shall be required to be initiated via pyrotechnics. Deployment telemetry shall be calibrated during this test.

4.5 ENVIRONMENTAL TEST MATRIX

The environmental test program shall be in accordance with Table 14.

APPENDIX A

SPECIFICATION FOR COMPUTER-CONTROLLED GROUND TEST SYSTEM

I. INTRODUCTION

The computer-controlled ground test system shall be provided to support all electrical tests and required launch support of each spacecraft and its instruments. This support shall include, but not be limited to, spacecraft instrument bus acceptance; integration and full-up system testing; thermal vacuum orbital testing; and launch, ascent, and early orbit activities as identified in this Specification. The functional hardware and software requirements, the associated quality, longevity, and maintenance and management requirements for the test system are presented in the following sections.

The required test system shall be similar to the Advanced Tiros-N Automatic Ground Equipment (ATNAGE) used for NOAA-HIJ in systems concept. The following are salient features of this system: (1) two relatively independent ground systems are included in the test complex, (2) general- and special-purpose test and telemetry equipment is interfaced with the computer bus, (3) an independent off-line processing capability is available, and (4) an automated test language is available and is used for converting test procedures to the required computer language for execution. Mod 500 CCR 1808

New features of the test system shall be selected to support the unique characteristics of the NOAA-N and N-prime spacecraft. The instruments shall ensure the availability of adequate processing and display capabilities and shall ensure that the test capability degenerates gracefully in the event of individual test hardware failure. Mod 527 CCR 1943

II GENERAL REQUIREMENTS

A. Tests Supported

The computer-controlled ground test system shall be designed to support all electrical tests defined in the test plan for integration and test (I&T) of the satellite, including orbital simulation, at the contractor's facility and all electrical systems tests and launch support at the Western Range (WR) launch site. The test system operation during orbital simulation shall simulate Spacecraft Operations Control Center's (SOCC) evaluation of the spacecraft and in particular, shall use normal telemetry instead of hardwire data, whenever possible.

In all cases, the test system shall be designed to provide the capability to receive, process, archive, retrieve, and display satellite data for verification of flight readiness or for the evaluation of spacecraft status and operation during launch, ascent, and post ascent operations to loss of signal (LOS) by WR and support aircraft.

B. Longevity and Growth

The hardware, software, documentation, training, and maintenance will be such that the test system will remain operational for a 10-year period, assuming: (1) continuous use for each spacecraft's test and launch support and (2) its shipment to and use at the WR to support each spacecraft launch, and its postlaunch return and installation in the contractor's test facility. The continuity required for remaining operational over this period shall be supported by the design, software and data storage and retrieval medium, documentation, maintenance spare parts availability, and other related factors.

The hardware and software will have a growth capability at least sufficient to accommodate the known satellite growth capability. The hardware, interconnection, and software documentation will be sufficient to support modifications or additions to accommodate this growth.

C. Quantity

There shall be a minimum of two independent test systems and one dedicated off-line archival, retrieval, processing, and display capability.

Each system shall be fully capable of supporting any phase of test or launch (i.e., WR operation) of a single spacecraft. The two systems shall be capable of operation concurrently to support the test and/or launch of two spacecraft. In addition, the systems shall be capable of being used as backups to one another. Parts, modules, and subsystems interchangeability between the systems will be a design requirement.

The off-line capability shall be provided at the contractor's facility or the WR, but not concurrently. Interchangeability with the test systems is desirable.

D. Control of Tests

Test operations shall be conducted and controlled through computers within the test systems (as used in this section, "test," is intended to include all WR launch operations). An automatic test language capability shall be provided that shall accept formatted English language test procedure statements (i.e., test scripts) and generate the associated sequence of machine language statements required to enable the computer to conduct the desired test. All software required to compile, assemble, edit, list, or otherwise implement the required language on the computers of the test system shall be provided.

The capability shall exist to interrupt any test at any time, to insert automatic test language commands through the operator's terminal, and to resume or terminate the test at the operator's option. The capability shall exist to restart any test at previously designated points within a test procedure after a failure or interruption to the procedure.

There shall be complete visibility of all tests. This visibility shall include display of the procedure step referenced to the complete procedure, the capability to list the procedure at any time except during execution of the procedure or alternately listing during execution of the procedure by another computer from the backup disk, and the capability to list information related to the procedure such as tables of constants, lists of sample points, and engineering unit conversions. In addition, there shall be an events log appearing on a printer and recorded on bulk storage in a form retrievable by the system. The events log shall show the state of the system, local and spacecraft time, any errors, the procedure run, and the commands sent to the spacecraft on an event-by-event basis, and will also log manual commands that may be inserted into the procedure. In addition to the procedure being run from the automatic test language, the capability will exist to monitor and check a basic set of data points not under the control of the main procedure. An independent mechanism for decoding and printing of the test system generated command stream shall be provided. The printed output shall provide spacecraft decoded commands versus ground time and shall be separate from the test system computer printout.

A Tiros data base system will be available. A satellite engineering description (telemetry/command descriptors, analog limits, analog engineering conversion constants, and a command/telemetry correspondence) shall be maintained separately on-line bulk storage. The data base will be input to the on-line test system prior to the test. This will facilitate software modification in that spacecraft parameters will not be software hard coded. There will be an interactive edit capability when the data base is transferred to use with each test.

E. Data Recording and Presentation

All real-time TIP data occurring during test shall be recorded in formatted form on digital computer tape. These tapes shall be saved until released by NASA. The capability shall exist to replay the data from these tapes into the system and perform any operation possible on the original data. The replay of these data shall be possible after the receipt of real-time TIP data or on the off-line system at any time after the digital tape is complete. Mod 500 CCR 1808

A mechanism to receive and record the direct 66.54-kbps global area coverage (GAC) data shall exist. (This implies a hardware interface to the spacecraft). A mechanism will be required for checking the 66.54-kbps GAC data directly to verify operation of that portion of the MIRP, as well as a mechanism to record these data for spacecraft tape-recorder testing. Suitable mechanisms will exist for controlling the recording and for correlating the playback to the spacecraft tape recorder dump data.

A mechanism shall exist to retrieve and check hardwired digital APT data. The operation of the APT MIRP algorithm shall be checked.

A mechanism shall exist to record the 665.4-kbps HRPT data for correlation with spacecraft tape recorder dump data, in addition to what may be necessary for AVHRR and MIRP testing.

A mechanism shall exist to retrieve the AMSU and TIP data from the HRPT data stream, format the data, and place it into the function file for processing.

The capability shall exist to test for interference of all data streams that may occur simultaneously. This testing does not necessarily have to be done in real time. An analog recorder is acceptable for storing the simultaneous transmissions for subsequent analysis, and this is also acceptable for storing spacecraft tape recorder dumps for subsequent checking. In addition, there will be a mechanism to provide digital tapes of the TIP data stream for both analysis of the data and training simulation. Mod 500 CCR 1808

F. Concurrency of Operation

The 8.32-kbps TIP data must be continuously presented for real-time computer analysis and recorded on digital or analog magnetic tapes. Application programs and any additional basic check program, which run independently of the automatic language procedure, must also have access to the real-time TIP data.

Concurrent with the TIP operations, real-time receipt, monitoring and display of the HRPT data (665.4 kbps), and the direct GAC (66.54 kbps) data are required together with any necessary recording of the data for either analysis or spacecraft tape recorder checks. Pictorial display of the HRPT data and the APT data is required concurrently with other operations; however, it is desirable that the picture display equipment be able to operate independently of the computer.

During interference checks, all data streams shall be stored simultaneously, concurrent with the above operations; however, the storage of spacecraft tape recorder dump data may be on an independent analog recorder for subsequent off-line testing.

Concurrent real-time analysis of the TIP spacecraft subsystem data shall include limit checking, command checking, and status checking. Concurrent real-time analysis of the HRPT and direct GAC data shall include sync word verification. Other appropriate types of information within each of the data streams shall be checked. It shall be a design goal to concurrently process all real-time Digital A data (radiometric data) contained in the TIP data.

The validation of instrument response shall require monitoring target operation; therefore, concurrent with previous operations, a data acquisition system must be controlled by the computer and used to measure target information and the information must be converted for use in validating the instrument response. Closed-loop control through the computer is not anticipated for target control; however, the system should alert the operator when targets are not functioning as expected.

G. Data Availability

The events list shall be available to the printer continuously; however, actual printing will be under operator control.

Recording of the events list will occur continuously during test; however, the system will be capable of retrieving the last 20 events messages without interrupting the recording of newer data.

A "snap" generator mechanism shall be available under the automated test language, under which a standard size format of information will be available, which may be conveniently filled from various data sources such as real-time data points, calculated data, and status data. The specific snaps made up under this general mechanism will be able to be put in a library from which they may be called through commands of the automated test language. The output of these snaps will be sent to cathode-ray tube (CRT) screens or to a printer. The printing of these snaps must not interfere with the events log.

Previously recorded data must be able to be retrieved, processed, and displayed. History data and events should be able to be retrieved from tape, operated on if desired, and all display capability present for real-time data should be available for display of the recorded data. In addition, short-term recorded data such as HRPT video data should be able to be read from tape and operated on by a background applications program.

H. Data Acquisition System

A data acquisition system controlled by the computers within the test system shall be required to read information from satellite simulators. It is expected that this is a low-sampling rate system and that a commercial multiplexer/digital voltmeter system will be sufficient.

I. GSFC Line Interface

Either ground test system (ATNAGE) shall be capable of sending TIP data to or receiving spacecraft TIP data from the Spacecraft Operations Control Center (SOCC) at NOAA, Suitland, MD via NASA NASCOM provided devices over a GSFC data line and modem at 9.6 kbps or 56 kbps from or to the contractor's Sunnyvale facility or the WTR. Mod 500 CCR 1808

J. Commanding

Either ground test system shall be capable of generating and transmitting the real-time and stored commands required to integrate, test, launch, and operate this series of spacecraft. The systems shall also be capable of confirming the spacecraft receipt and execution of all commands using OBC memory dumps, TIP telemetry, and related data.

K. Self-test

Self-test hardware and software shall be included as part of the system. This self-test has two purposes: (1) to provide a hardware and software acceptance test prior to first use with the spacecraft, and (2) to provide routine self-test and diagnostic capability during the life of the equipment. It is an objective to completely debug the hardware and software prior to the first testing with the spacecraft so that the system can effectively aid in determining first integration problems. Therefore, high confidence that the test system is operating correctly at the start of integration requires good acceptance testing of the test system hardware and software.

III. SOFTWARE DEFINITION

A. The software will provide the functional capabilities described in IS-2295960, ATNAGE Requirements.

B. The required software shall also include a Statistical Analysis and Graphic Display System capable of operationally:

a) accepting and archiving all properly formatted data produced by the TIROS-K through N’ spacecraft, associated instruments and ancillary control data from the LMAS Integration and Test facility, the various instrument manufacturers, the Western Range launch facility, the NOAA SOCC, and other approved contributors

b) retrieving specified data from this archival database and preparing the retrieved data for subsequent processing (i.e., for display, for analysis, for dissemination, etc.)

c) processing and/or displaying raw and/or processed data from the archived database, as required to effectively evaluate the spacecraft and instuments in timely fashion, and distributing this data in the formats required to approved users by hard copy, digital tape, cassette tape, floppy disc, modem, and/or selected network.

The system shall be capable of operationally accepting, archiving, retrieving, processing, and distributing properly formatted processed data from approved sources (i.e., individual data analyst at LMAS, instrument manufacturer, SOCC, etc.).

a) the on-line archival of not less than 30 days of spacecraft, instrument, and ancillary data;

b) the off-line archival of all available spacecraft, instrument, and ancillary data;

c) the retrieval of selected on-line archived data in near-time; and

d) the retrieval of selected off-line archived data within 30 minutes.

The System shall be user-friendly and menu driven, and the design shall be open-ended to facilitate the inclusion of additional data edit, analysis or distribution algorithms, additional data types, additional user, etc.

IV. TEST PLAN

The computer-controlled ground test system will be tested and accepted based on the following general test plan. The definition of the components appears in PS-2285033, Specification for Tiros-N Electrical Aerospace Ground Equipment (NAGE).

A. General

1. All tests performed on the ATNAGE will be performed under the cognizance and direction of a responsible engineer.

2. Any deviations or anomalies occurring during a test will be documented. Any corrective action taken or required as a result of each anomaly will be documented.

3. The tests to be performed are as follows:

   - Power-up Box Electrical Test**
   - Ordnance Device Simulator Electrical Test**
   - SAGE Functional Check (2)
   - Spacecraft Simulator (SCS) Functional Check
   - SSR Functional Check (2)
   - System-Software System Functional Test
   - System Performance Test (2)**

4. Each test will be performed in room ambient conditions in accordance with an approved test procedure.

** Successful completion of test will demonstrate compliance with 2280533, Section 3.

B. Power-up Box Electrical Test

This test will verify that the Power-Up Box (PUB) can connect the bus batteries into the bus power subsystem under manual control of the operator. The ability to accurately monitor bus voltage and current test points during and after the power-up sequence will be verified.

C. Ordnance Device Simulator Electrical Test

This test will verify that the ordnance device simulator (ODS) accurately simulates each ordnance device in the spacecraft. The impedance of each simulator will be veri-fied. The activation of each simulator at minimum device all-fire current will be verified.

D. SAGE Functional Check

This will use special software drivers to verify the proper functional operation of each interface between the computers/software and peripherals. The ability to output data to display devices (CRTs, teletype message (TTY), printers, and strip charts) will be verified.

E. SCS Functional Check

This test will use the spacecraft simulator software system to verify the proper functional operation of each interface between the computer/software and external hardware. The ability to generate correctly formatted simulation of spacecraft data outputs will be verified.

F. SSR Functional Check

This test will use the bus software system, one Stationary Aerospace Ground Equipment (SAGE), and hardware input simulators to verify the proper functional operation of the interface between the bus software system and the signal scanning and measurement devices in the Spacecraft Support Rack. The ability to program the correct channel and to transfer accurate measurements will be verified.

G. System-Software Functional Test

This test will use the bus software system, the spacecraft simulator system, special test software procedures, and one SAGE to verify the proper functional operation of the integrated software and its interfaces with system input/output devices and test display devices in a closed-loop mode (no spacecraft or SSR data inputs).

H. Dual Functional Test

This test will verify the capability of the ATNAGE system to transmit commands to and receive TIP data from two spacecraft buses concurrently.

I. System Performance Test

This test will use all software systems, one SAGE, one SSR, and SSR hardware input simulators to verify AGE system performance. The spacecraft simulator system will be used to simulate spacecraft command and telemetry data interfaces. The following performance parameter will be verified:

- System software loading
- Bus data base generation
- Atlas program compilation
- Storage and retrieval of digital or analog playback data

- Control of all programmable devices and interfaces
- Initiation and control of system software operation****
- Initiation and control of Atlas test programs****
- Transmission and Verification of spacecraft commands***
- Spacecraft memory loading, dump, and verification***
- Processing of TIP data in boost, orbit, dwell, and CPU dump formats nonsimultaneously
- Monitoring of spacecraft power switching and hardware status signals***
- Storage of digital data during real-time operations and off-line retrieval of the data
- Output of data to test displays under Atlas, system software, or operator control***
- Spacecraft and STE hardline interfaces***

*** Concurrently

**** Control, not initiation, to proceed concurrent with other functions.

APPENDIX B

ACRONYM LIST

ADACS	Attitude Determination and Control System
ADE	Array Drive Electronics
ADP	Automatic Data Processing
AF	Air Force
AGE	Aerospace-Ground Equipment
AGS	Ascent Guidance Software
AH	Ampere-Hour
AIP	AMSU Information Processor
AKM	Apogee Kick Motor
ALP	Ascent Load Package
AM	Amplitude Modulation
AMSU	Advanced Microwave Sounding Unit
APL	Approved Parts List
APT	Automatic Picture Transmission
ARIA	Advanced Range-Instrumentation Aircraft
ARO	After Receipt of Order
ASD	Astro Space Division
Atlas	Abbreviated Test Language for Air Systems
ATN	Advanced Tiros-N
ATNAGE	Advanced Tiros-N Aerospace Ground Equipment
AVE	Aerospace-Vehicle Equipment
AVHRR	Advanced Very High Resolution Radiometer
BA	Beacon Antenna
BCA	Battery Charge Assembly
BLP	Backup Load Package
bps	bits per second
BTX	Beacon Transmitter
BVR	Boost Voltage Regulator
C&CS	Command and Control System
CAP	Controlled Access Period
CCR	Configuration Change Request
CD	Command Diplexer
CDA	Command and Data Acquisition
CDR	Critical Design Review
CDU	Command Decoding Unit
c.g.	Center of Gravity
CIU	Controls Interface Unit
CM	Configuration Management
CP	Computer Program
CPC	Controls Power Converter
CPU	Central Processing Unit
CRT	Cathode-Ray Tube
CXU	Command Annex Unit
DACS	Data Acquisition and Control System
DAU	Description/Authenticating Unit
dB	Decibel
dBi	Decibel Isotropic
dBm	Decibel Above 1 Milliwatt

DCAS	Defense Contract Administrative Service
DCS	Data Collection System
DET	Direct Energy Transfer or Detailed Electrical Test
DHS	Data-Handling Subsystem
DMSP	Defense Meteorological Satellite Program
DPD	DCS Processor Diplexer
DPSS	Data Processing Services System
DPU	Data Processing Unit
DTR	Digital Tape Recorder
DTSR	Dual-Test Support Rack
DTT	Direct TIP Transmission
EAGE	Electrical Aerospace Ground Equipment
ECN	Engineering Change Notice
EIRP	Effective Isotropic Radiated Power
ELT	Emergency Locator Transmitter
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EP&DS	Electric Power and Distribution Subsystem
EPIRB	Emergency Position Indicating Radio Beacon
ESA	Earth Sensor Assembly
ESM	Equipment-Support Module
FCP	Flight Computer Program
FGGE	First GARP Global Experiment
FM	Frequency Modulation
FOV	Field of View
FSA	Filter and Switch Assemblies
FSE	Factory-Support Equipment
FSK	Frequency Shift-Keyed
FSSS	Flight Software Support Software
GAC	Global Area Coverage
GARP	Global Atmospheric Research Program
GDC	General Dynamics Convair
GE	General Electric
GEOS	Geophysical Orbiting Satellite
GFE	Government-Furnished Equipment
GFP	Government-Furnished Property
GN_2	Gaseous Nitrogen
GOES	Geostationary Operational Environmental Satellite
GPS	Global Positioning System
GRD	Ground STDN Receiver Demodulator
GS	Ground Station
GSE	Ground-Support Equipment
GSFC	Goddard Space Flight Center
GSSS	Ground-Support Software System
HEPAD	High Energy Proton and Alpha Detector
HIRS	High Resolution Infrared Sounder
HRPT	High Rate Picture Transmission

I&T	Integration and Test
IMP	Instrument Mounting Platform
IMU	Inertial Measurement Unit
IR	Infrared
I/O	Input/Output
ISS	Integrated Spacecraft System
LAC	Local Area Coverage
LOS	Loss of Signal
LUT	Local User Terminal
LVS	Launch-Vehicle Systems
mb	Millibar
MEPED	Medium Energy Proton and Electron Detector
MIRP	Manipulated Information Rate Processor
MHS	Microwave Humidity Sounder
MSDF	Multisoftware Development Facility
N_2	Nitrogen
N_2H_4	Hydrazine
NAGE	NOAA Aerospace Ground Equipment
NASA	National Aeronautics and Space Administration
NESDIS	National Environmental Satellite, Data, and Information Service
nml	Nautical Miles
NOAA	National Oceanic and Atmospheric Administration
NOMSS	National Operational Meteorological Satellite System
NRZ	Nonreturn to zero
OBC	Onboard Computer
ODS	Ordnance Device Simulator
PCM	Pulse-Code Modulation
PLC	Propellant Loading Cart
PM	Phase Modulation
PMICS	Program Management Information Control System
PMP	Parts, Materials, and Processes
POPS	Power-on Processor Software
PSE	Power-Supply Electronics
psia	Pound per square inch absolute
PTC	Pitch Torque Coils
PUB	Power-Up Box
Q	Magnification Factor
QC	Quality Control
QCM	Quality Control Monitor or Quartz Crystal Microbalance
RAMS	Random-Access Measurement System
R&QA	Reliability and Quality Assurance
RAU	Remote Acquisition Unit
RCE	Reaction Control Equipment
RCPE	Reaction Control and Propulsion Equipment
RCS	Reaction Control Subsystem
RF	Radio Frequency

RFF	Radio Frequency Filter
RFI	Radio-Frequency Interference
RFS	RF Switch
RM	Rocket Motor
rms	root mean square
RSS	RCE Support Structure
RWA	Reaction-Wheel Assembly
RXO	Redundant Crystal Oscillator
RYC	Roll/Yaw Coil
SA	Solar Array
SAD	Solar-Array Drive
SAGE	Stationary Aerospace Ground Equipment
SAR	Search and Rescue
SARP	Search and Rescue Processor
SARR	Search and Rescue Repeater
SAS	Solar-Array Support
SBA	S-band Antenna
SBW	Solar Backscatter Ultraviolet Radiometer
S/C	Spacecraft
SCF	Satellite Control Facility
SCO	Subcarrier Oscillator
SCU	Signal Conditioning Unit
SE	Support Equipment
SEL	Space Environment Laboratory
SEM	Space Environment Monitor
SEPET	System Electrical Performance Test
SGLS	Space-to-Ground Link Subsystem
SLA	SARR L-band Antenna
SLP	Standby Load Package
SMP	Software Management Plan
SMS	Small Meteorological Satellite or Synchronous Meteorological Satellite
S/N	Signal to Noise
SCU	Signal Conditioning Unit
SOA	S-band Omniantenna
SOCC	Satellite Operations Control Center
SOT	Spacecraft Operational Test
SRA	SARR Receiver Antenna
SSD	Sun Sensor Detector
SSE	Sun Sensor Electronics
SSR	Solid State Recorder
SSR	Spacecraft Support Rack
STDN	Space Tracking and Data Network
STS	Space Transportation System or Structure Subsystem
STX	S-band Data Transmitter
TCE	Thermal-Control Electronics
TCS	Thermal-Control Subsystem
TDR	Test Discrepancy Report
TE	Test Equipment
TED	Total-Energy Detector
TIP	Tiros-N Information Processor
TIROS	Television Infrared Observation Satellite
TLM	Telemetry

TOVS	Tiros Operational Vertical Sounder
TRB	Test Review Board
TSS	Tiros Spacecraft Simulator
TTY	Teletype Message
T/V	Thermal Vacuum
UDA	Ultrahigh Frequency Data Collection System Antenna
UIIS	Unique Instrument Interface Specification
ULP	Unified Load Package
USAF	United States Air Force
VCM	Volatile Condensable Material
VTX	VHF Transmitter
WSMC	Western Space and Missile Center
XPOPS	Extended Power-on Processor Software
XSU	Cross Strap Unit

APPENDIX C

WAIVERS/DEVIATIONS

Deviation/ Waiver No.	CCR No.	CCR Approved Date	Section/ Ref. Doc./ Effectivity	Description
	2339	05/07/97	4.4.3, 4.5 (Table 14) L	LMMS-EWO is requesting a waiver for the EMI/EMC performance of the Decryption Authentication Unit (DAU), part number 3287854-501, serial number M315001. The DAU exceeds the narrow band radiated emissions requirements as specified in MIL-STD-461C RE02. Specifically, the DAU exceeds MIL-STD-461C specifications at 5 discrete frequencies by the amounts listed below: 23.71 MHz +4.0 dBuV/m 24.57 MHz +6.8 dBuV/m 25.77 MHz +3.3 dBuV/m 26.70 MHz +5.7 dBuV/m 27.67 MHz +0.8 dBuV/m These levels were found to occur with the STE and DAU both powered "on".
	2340	05/07/97	4.4.3, 4.5 (Table 14) L	LMMS-EWO is requesting a waiver for the EMI/EMC performance of the Power Supply electronics (PSE), part number 3281154-501, serial number M300001, and the Battery Charge Assembly (BCA), part number 2633667-502, serial number F000011. The PSE and BCA exceed the broadband and narrowband radiated emissions requirements as specified in MIL-STD-461C RE02. Specifically, the PSE and BCA exceed MIL-STD461C in the following areas, at the nominal frequencies and amounts listed below: Narrowband Radiated Emissions (RE02): Wideband Radiated Emissions (RE03): 28.0 MHz 49.0 dBuV/m 25.8 MHz 76.3 dBuV/m 25.8 MHz 50.0 dBuV/m
	2342	05/19/97	3.6.1.5 (k) & (l) KLMNN’	LMMS-EWO is requesting a waiver for the performances of the Communications Subsystem Antennae while mounted on the spacecraft. The non-conformances were found during the Antenna Model Measurements and documented in 2 reports referenced in CCR. UHF DCS/SARP Receive Antenna (UDA): At 401.6 MHz, 0.2% of the radiation field is below the min. gain spec. and 33.7% is above the max. gain spec. At 406.0 MHz, 3.7% of the radiation field is below the min. gain spec. and 78.9% is above the max. gain spec. SRA: At 121.7 MHz, 10.3 % of the radiation field is below the min. gain spec. and 19.7% is above the max. gain spec. At 243.5 MHz, 99.2% of the radiation field is above the max. gain spec. At 406.0 MHz, 0.8% of the radiation field is below the min. gain spec. and 85.1% is above the max. gain spec. SLA: At 1544.5 MHz, 17.0% of the radiation field is below the gain spec. and 30.2% of the radiation field exceeds the ellipticity spec.

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				SBA: At 1698.0 MHz (SBA-1), 2.4% of the radiation field is below the gain spec. and 4.0% of the radiation field exceeds the ellipticity spec. At 1702.5 MHz (SBA-2), 6.6% of the radiation field is below the gain spec. and 3.5% of the radiation field exceeds the ellipticity spec. At 1707.0 MHz (SBA-3), 3.7% of the radiation field is below the gain spec. and 4.1% of the radiation field exceeds the ellipticity spec. VRA: At 137.5 MHz, 37.0% of the radiation field is below the gain spec. and most of the radiation field exceeds the ellipticity spec. SOA: At 2247.5 MHz (SOA-3) and SOA-4 as a pair), 1.0% of the radiation field is below the full sphere coverage gain spec.
	2338 A	06/18/97	3.6.1.8, 4.4.8 KLM	This waiver is to accept a 0-500 PSI range hydrazine telemetry pressure transducer in lieu of the subsystem specified 0-2500 PSI range transducer for the TIROS NOAA-KLM spacecraft.
	2348	06/18/97	4.4.7 K	Waiver is for deficient TCE-related set point data for the ATN-K spacecraft. Request that complete T/V test, TCE set point verification for this spacecraft (NOAA-K) not be required. During s/c-K T/V testing, several TCE-related set points were missed, or their measured values were out of limits. The Performance Spec. requirement states that "...During the test, all active and passive louver set-points shall be verified. All heater actuation set-points shall be verified for all heater controllers..."
	2341	07/28/97	3.6.1.5(q) MNN'	LMMS-EWO is requesting a waiver for the performances of the SLAs, part number 2297967-501, serial #s Y270020, Y270021, Y270022, Y270023, and Y270024. The SLAs do not meet the Perf. Spec. requirements as follows: Serial # Performance Deficiencies Y270020 Gain is nominally 0.8dB below spec. over the region 0 deg.<theta<60 deg, & the axial ratio exceeds the spec. by 1.0dB, max. Y270021 Gain is nominally 1.0dB below spec. over the region 0 deg.<theta<60 deg., the axial ratio exceeds the spec. by 1.1dB, max, and the return loss exceeds the spec. by 1.9dB. Y270022 Gain is nominally 1.1dB below spec. over the region 0 deg.<theta<45 deg. Y270023 Gain is nominally 0.9dB below spec. over the region 30 deg.<theta<60 deg. Y270024 Gain is nominally 1.1dB below spec. over the region 0 deg.<theta<45 deg., & the axial ratio exceeds the spec. by 1.5 dB, max.

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	2345	08/06/97	4.4.7 KLM	Waive NOAA-KLM requirement that min. temperature extreme for RSS during s/c flight acceptance T/V testing be its min. predicted temperature (-48 deg.C) less 5 deg. C for NOAA-K only. Unable to achieve a –53 deg. C RSS temperature during the cold plateaus of two NOAA-K T/V tests. Cold plateau RSS temperatures for the first test were: -31 deg. C min. –25 deg. C max. and –28 deg. C average. Temperatures for the 2nd test and the NOAA-L T/V test were: -42 deg. C min., -34 deg. C max. and –38 deg. C average. *Note: Waiver was approved for K&L requirements (-53 deg. and –48 deg. respectively) @ -42 deg. A deviation was approved for NOAA-M requirement (-48 deg.) @ -42 deg. (level achieved on NOAA-K and –L).*
	2371	04/14/99	3.6.1.5(e) LMNN’	LMMS is waiving the VHF Beacon Transmitter BTX bit asymmetry requirement from 3% to 10%. S/C — S/N — Measured Bit Asymmetry — Spec. Limit N/N’ — 615 — 9.64% — <3% N/N’ — 616 — 7.64% — <3% N/N’ — 617 — 5.32% — <3% N/N’ — 618 — 8.0% — <3% N/N’ — 619 — 7.98% — <3% N/N’ — 620 — 6% — <3% K — 511 — 3.5% — <3% K — 614 — 4.0% — <3% L — 510 — 6.0% — <3% L — 613 — 5.6% — <3% M — 509 — 4.4% — <3% M — 612 — 6.0% — <3%
	2368 A	05/14/99	3.6.1.5(r) M	Waiver is for the level of transmitter spurs at the SARR, DCS, and SARP receivers input. The non-conformances were found during the NOAA-M EMI test held on February of 1998. The spurs are not in the passband of the receivers and will not impact the operations of the receivers. A summary of the discrepancies is as follows: SARR 406 MHz Receiver: From 407.20 MHz to 407.24 MHz, max. level of spurs is -115.4dBm, spec. limit is <-125dBm. Spurs caused by mixing of STX2, STX4, and BTX2 fundamental frequencies. From 412.16 MHz to 412.36 MHz, max. level of spurs is -92.2 dBm, spec. limit is <-100 dBm. Spurs caused by BTX1 and VTX1 harmonics.

				From 413.15 MHz to 413.34 MHz, max. level of spurs is -89.7 dBm, spec. limit is <-100 dBm. spurs caused by BTX2 and VTX2 harmonics. SARP Receiver: From 412.358 MHz to 412.52 MHz, max. level of spurs is -97.8 dBm, spec. limit is <-100 dBm. spurs caused by BTX1 and VTX1 harmonics. From 413.00 MHz to 413.02 MHz, max. level of spurs is -94.4 dBm. spec. limit is <-100 dBm. spurs caused by BTX2 and VTX2 harmonics. DCS Receiver: At 402.7292 MHz, level of spur is -124.2 dBm, spec. limit is <-125 dBm. spur caused by mixing of STX4, STX3, and BTX2 fundamental frequencies.
	2383A	05/25/99	4.2.1.5 LMNN’	LMMS is requesting a deviation from the requirements of SBAs (P/N 20038448) protoflight/acceptance level thermal cycling test as stated in the S-480-25.a, S-480-26.1, LMMS 2629668, PN-2629891, and PS-20038448. The proposed SBA protoflight/acceptance level thermal cycling profile is summarized in ccr section 1. Test Para. — Current Req. — Proposed Dev. Cumulative Test Time (min.) — 32 hours (@ea. temp. extreme) — 16 hours (@ ea. temp. extreme)
	2369A	06/10/99	4.4.7 L	Waiver is for NOAA-L spacecraft heater and louver set-point measurements during T/V tests. These set points could not be met. The s/c level test method of verification is a qualitative visual reference that the louvers are operating. The pass-fail limits in the LMMS test procedure cannot be met for the TCE measurements (included in CCR) because the test is not designed to get the sensors cold enough.
	2372B	03/17/00	2598865 (3.1.1.1.2.1.5) L	Requirement 3.1.1.1.2.1.5 (2598865) limits the transient load currents drawn by units to 150 percent of the maximum average steady-state current drawn by the unit on the +28V bus, exclusive of turn-on. Since the maximum average steady-state current drawn by the SSA is 30 mA, the transient load current limit for the SSA is 45 mA. The SSE serial numbers 305 and 306, part number 2598865, exceed the allowable command enable current transient of 45 mA on the +28V bus. Up to 48 mA transients have been observed. Ref. DR W23648 and DR W21514. LMMS requests a waiver from the requirement that the +28V transient load current cannot exceed 150% of the maximum average steady-state current (in this case, 45 mA).

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	2398	04/20/00	2629668 (3.1.1.2.1.5) M	Requirement 3.1.1.2.1.5 (2629668) limits the transient load currents drawn by units to 150 percent of the max average steady-state current drawn by the unit on the +28V bus, exclusive of turn-on. Since the maximum average steady-state current limit drawn by the SSA is 30mA, the transient load current limit for the SSA is 45mA. The SSE serial number 306, part number 2598865, exceeds the allowable command enable current transient of 45mA on the +28V bus. A 46mA transient was observed. Reference DR W21514.
N/A	2387A	06/11/00	2629668, 3.1.1.2.1.5	Requirement limits the transient load currents drawn by units to 150 percent of the maximum average steady-state current drawn by the unit on the +28 V bus, exclusive of turn on. Since the maximum average steady-state current drawn by the SSA is 30 mA, the transient load current limit for the SSA is 45 mA.
N/A	2400	07/10/00	2629668, 4.6.1.7.2	Requirement limits the maximum signal level during EMI testing to –145 dBm for frequencies between 405.900 and 406.000 MHz. MIMU SN/ 74 on IMS S/N 102 had a spur at 405.9974 MHz that exceeded the –145 dBm limit by 2 dB. The spur occurred in the 50 Hz mode with vertical polarization.
N/A	2411BR1	03/04/03	2629668	The IMS S/N 102 MIMU-1 and MIMU-2 exceed Class 1 current ripple requirements. The requirements and the waived values are as follows:

	IMS Ripple Current		
	28 v Main Bus	10 volt I/F Bus	10 volt Serial I/O Bus
Class 1 Requirement IMS Specification 20058260	30 mA p-p max 55 mA p-p max	1 mA p-p max 1 mA p-p max	<1 mA p-p max 4 mA p-p max
Waiver Values	60 mA p-p max	6 mA p-p max	16 mA p-p max
IMS s/n 102 (NOAA-N) Highest Measured Value:			
- while on NOAA-N :	59.6 mA p-p	2.4 mA p-p	12.4 mA p-p

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N/A	2424R1	08/11/03	Paragraph 3.6.1.7.g	1.33 MHz split phase playback of GAC and LAC data for XSU model 2284379-505 is not compliant with specifications. Non-compliant conditions may be expected for: a) XSU Side 1 and playback from DDR4 or DDR5, data fails b) XSU Side 2 and playback from DDR1, DDR2, or DDR3, data fails There is no data loss to the mission with judicious selection of data path combinations.